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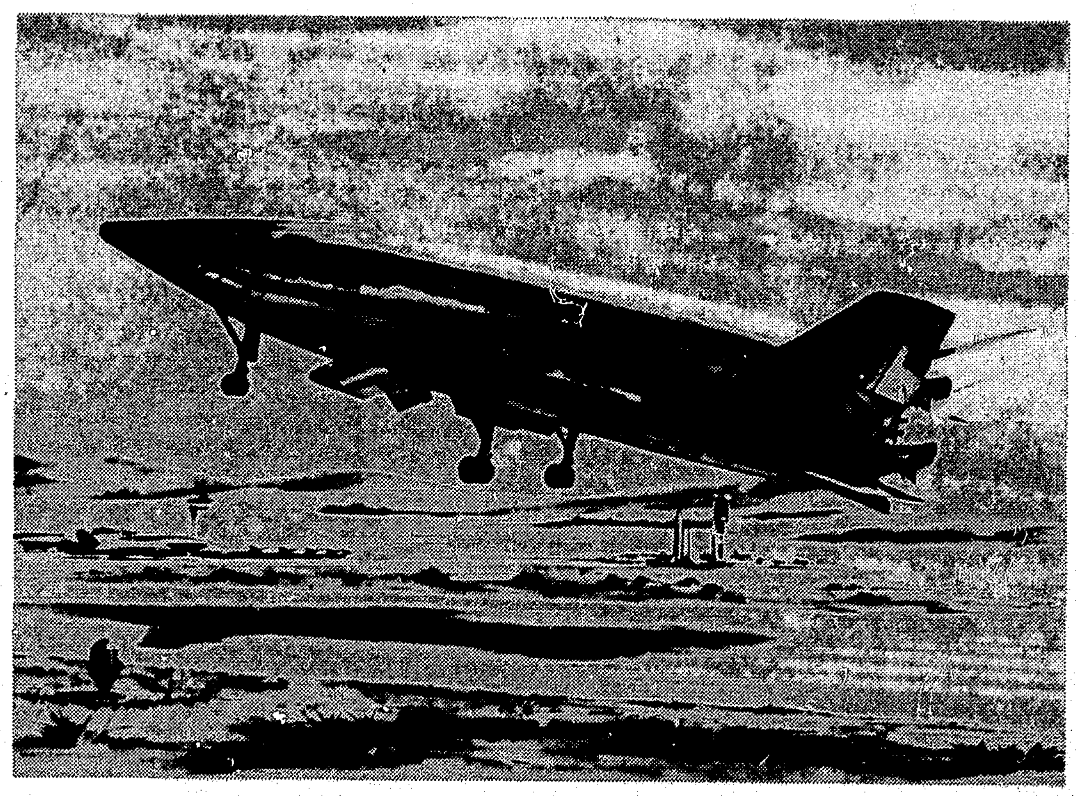

SPACE SHUTTLE FINAL TECHNICAL REPORT

VOLUME VIII ✦ MISSION/PAYLOAD AND SAFETY/ABORT ANALYSES

GENERAL DYNAMICS
Convair Division

054

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VOLUME VIII ✦ MISSION/PAYLOAD AND SAFETY/ABORT ANALYSES

31 October 1969

Prepared by
CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

FOREWORD

This volume of Convair Report No. GDC-DCB 69-046 constitutes a portion of the final report for the "Study of Integral Launch and Reentry Vehicles." The study was conducted by Convair, a division of General Dynamics Corporation, for National Aeronautics and Space Administration George C. Marshall Space Flight Center under Contract NAS 9-9207 Modification 2.

The final report is published in ten volumes:

Volume I	Condensed Summary
Volume II	Final Vehicle Configurations
Volume III	Initial Vehicle Spectrum and Parametric Excursions
Volume IV	Technical Analysis and Performance
Volume V	Subsystems and Weight Analysis
Volume VI	Propulsion Analysis and Tradeoffs
Volume VII	Integrated Electronics
Volume VIII	Mission/Payload and Safety/Abort Analyses
Volume IX	Ground Turnaround Operations and Facility Requirements
Volume X	Program Development, Cost Analysis, and Technology Requirements

Convair gratefully acknowledges the cooperation of the many agencies and companies that provided technical assistance during this study:

NASA-MSFC	Aerojet-General Corporation
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NASA-ERC	Pratt and Whitney
NASA-LaRC	Pan American World Airways

The study was managed and supervised by Glenn Karel, Study Manager, C. P. Plummer, Principal Configuration Designer, and Carl E. Crone, Principal Program Analyst (all of Convair) under the direction of Charles M. Akridge and Alfred J. Finzel, NASA study co-managers.

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SECTION 1

INTRODUCTION

This volume contains the results of two ILRV study areas: mission/payload analysis and safety/abort analysis. Sections 1.1 and 1.2 present a statement of the problem, the approach to its solution, and a summary of significant results for each of the above study areas.

1.1 MISSION/PAYLOAD ANALYSIS

The mission/payload analysis, a special emphasis task, addressed the problems of determining the space shuttle payload capability for the baseline and alternate missions and the utilization of this payload capability.

Functional requirements must be developed for each mission and converted to design concepts. Development of both payload and deployment/retrieval concepts will allow an examination of the interaction with specific mission interface requirements and the recognition of commonality among missions. A space shuttle mission traffic schedule and baseline and alternate mission descriptions are first defined with the primary emphasis on the baseline space base/station logistics mission. Detailed main propulsion subsystem and attitude control subsystem ΔV requirements are developed for an optimized mission profile. Alternate mission performance sensitivity ground rules are discussed, mission profiles are presented, and FR-1 alternate mission payload capability is assessed. A modular payload concept is developed to satisfy all missions with a minimum number of module configurations. The interaction of docking and payload operations with the payload design is analyzed as is the influence of all missions on the final determination of payload module design and on-orbit operations. Preliminary module designs are developed and key tradeoff studies are performed.

The FR-1 performance capability is summarized on Table 1-1. Payloads range from 0 to 88,600 pounds for on-orbit ΔV requirements of 4145 to 1010 fps, respectively for the FR-1 configuration. The mission ΔV requirements presented are also applicable to FR-3 and FR-4. Similar alternate mission capability exists for FR-3 and FR-4. The study has shown that the space shuttle system is compatible with the missions identified in the NASA space shuttle task group report. In some instances, modifications to the basic orbiter may be required for specific missions; however, the modifications can be minimized if a modular payload approach is used.

It was concluded that a modular payload approach is feasible and desirable for the space shuttle system. This approach minimizes performance penalties to the basic

Table 1-1. FR-1 Mission Performance Summary

Mission	Orbit Altitude (n.mi.)	Orbit Inclination (deg)	On-Orbit ΔV (fps)	Payload Capability (lb up)
Space Base/Station Logistics	270	55	1800	50,000
Delivery of Propulsive Stage and Payload	100	28.5	1010	88,600
	200	55	1300	65,400
Placement, Retrieval, Service, and Maintenance of Satellites	100	28.5	1010/2040	88,600/56,000
	800	90	4010/4145	0/0*
Delivery of Propellants	200	28.5	1450	74,900
	300	55	1810	46,600
Short-Duration Orbit	100	28.5	1010	88,600
	300	90	1820	21,800
Space Rescue	270	55	3990	0

*Service, Maintenance, or Retrieve

space shuttle system because mission peculiar equipment will not be carried on alternate missions. However, two exceptions to this general rule appear justified:

a. The cab of the orbiter should have provisions for four mission related personnel. The 6-man orbiter cab would be used for 94 percent of the missions and would eliminate the need for a small personnel module.

b. The baseline vehicle propellant tanks should be sized to include on-orbit ΔV requirements of the large traffic missions (space station and propellant delivery). This ΔV requirement is estimated at 1800 ft/sec. This will eliminate the requirement to add or change tanks on alternate missions, which affects turn-around time and reliability. The tanks could be off-loaded on missions not requiring this much orbit ΔV (delivery of propulsive stages, etc.).

   ACPS propellant provisions for a ΔV of 200 fps should also be included for all missions with the exception of extended orbital missions where the orbiter is used as a laboratory base or for Earth surveillance. ACPS thruster arrangements should be compatible with docking requirements (control of translation and rotation).

1.1.1 SPACE BASE/STATION LOGISTICS. A 12-man personnel/cargo module was designed to verify module size and weight. This module can be converted to a cargo module.

Docking and orbital operations lead to the conclusion that a universal docking mechanism should be used on the baseline space shuttle.

1.1.2 DELIVERY OF PROPULSIVE STAGE AND PAYLOADS. It was concluded that the baseline space shuttle, when augmented by a propulsive stage using LH_2/LO_2 engines, can delivery payloads to synchronous orbit altitude with the useful payload weight being a function of the mode of operation. The study also identified fluid, mechanical, and structural interfaces between the two vehicles.

1.1.3 SATELLITE DELIVERY, MAINTENANCE, AND RETRIEVAL. The recommended baseline for in-orbit maintenance is a separate pressurized work module. Retrieval of inoperative satellites presents unique problems due to random tumbling. A stabilization and maneuvering system is proposed as a solution. This maneuvering pack would be independent of the satellite's operational system, and could be controlled by an orbiter operator.

1.1.4 PROPELLANT DELIVERY. A preliminary evaluation of alternate propellant delivery concepts indicates that the space shuttle should transfer a full tank rather than use a propellant transfer system. This will minimize losses associated with in-orbit fluid transfer and will reduce the number of space shuttle flights required.

It is also concluded that LH_2 delivery is volume limited by the baseline (15 ft × 60 ft) payload bay.

1.2 SAFETY/ABORT ANALYSIS

The requirement for a safety and abort analysis is based on the need to show a high probability of mission success and also a higher probability of successful intact abort. Safety and cost for space shuttles are the real drivers leading to requirements for a high probability of successful abort. Crew, passengers, payload, and the vehicles must be returned intact to make the space shuttle economically attractive. A safety and cost analysis conducted for the space shuttle was accomplished on the basis of intact abort. This analysis established safety and mission success goals which were used as a guide for the remainder of the study. Basic questions which were then addressed were:

What makes the space shuttle unsafe?

What action must be taken to change an unsafe situation into a routine abort operation?

How is safety improved?

What are the interfaces of safety with weight, operations, and mission success?

These questions were answered by conducting a gross failure and mission termination analysis with consideration given to:

a. Probability of occurrence of failures of subsystems, propulsion systems, and structure during the mission.

b. Abort options following these failures.

c. Availability of landing sites for aborted flights for several launch azimuths.

d. The fire and explosion hazard potentials of the stored propellants.

e. Intact abort, redundancy, and escape.

This analysis leads to:

a. Definition of abort procedures from the various mission phases.

b. Design requirements for the vehicle.

c. Design requirements for the minimization of the fire and explosion hazard.

The effect of engines on safety and mission success was studied using the probable range of engine reliabilities and the application of fail-operational and fail-safe criteria to both the booster and orbiter engines.

Finally, the safety, abort, and mission success characteristics of the FR-3 and FR-4 vehicles were defined.

Safety (probability of success of intact abort) and probability of mission success goals were established considering safety and cost effects as follows:

Safety	0.999 (1 loss/1000 flights)
Mission	0.97 (30 aborts/1000 flights)

The analysis shows that these goals can be approached or met by fail-operational/fail-safe and fail-safe design approaches and by minimizing fire and explosion hazards. Personnel safety in flight operations of the FR-3 and FR-4 vehicle concepts is achieved through intact abort. Safety and cost require that the crew, passengers, payload, and vehicle be returned intact following failures requiring abort. The safety and abort analyses show that intact abort is a feasible approach. The basic approach for treating the majority of failures is to provide redundancy to produce a fail-safe system or a fail-operational system. For the FR-3 and FR-4 vehicles, mechanical/electrical subsystems have fail-operational, fail-safe characteristics and the integrated avionics system has fail-operational, fail operational, fail-safe characteristics. When a failure occurs in a system with fail-operational characteristics there is no abort since the mission can be completed. When a failure occurs in a system at the fail-safe level it is necessary to go to an abort procedure.

The once-around abort procedure reflects action to achieve a high probability of successful intact abort from all failures. For failures during liftoff to staging phase

the booster and orbiter elements will complete the boost phase and stage when the booster propellants are depleted. The vehicles separate and the booster returns to the launch site in a normal manner while the orbiter continues once around the Earth and returns to the launch site.

There are failure situations which may require immediate abort. Typical of these types of failures are structural failure, thermal protection subsystem failure, and catastrophic situations such as a fire requiring early separation. Following separation, a throttle/burn-dump/reverse flight operation can be used. All remaining propellant is expended through the rocket nozzles and a flight profile is selected to allow the vehicles to return to the launch site.

For failures after staging, the orbiter abort mode is to continue once around the Earth and return to the launch site.

The FR-3 vehicle, with a 15-3 booster-orbiter engine arrangement, has a relatively low number of mission aborts because it incorporates fail-operational/fail-safe provisions for engines in the booster. The FR-3 can achieve staging with one engine out because the 7% overthrust capability of the booster engines allows the performance thrust-weight ratio to be maintained. The FR-3 can achieve intact abort with two engines out at lift-off. During the booster phase more engines can be out and intact abort is still possible.

The FR-3 and the FR-4 orbiter engines do not have fail-operational/fail-safe capability during the staging to orbit phase because the weight penalty to provide a 50% overthrust in the three-engine orbiter is prohibitive. The FR-3 and FR-4 orbiters do have fail-operational/fail-safe capability for all on-orbit maneuvers.

The FR-4,with a 9-3-9 booster-orbiter-booster engine arrangement, can achieve staging with one engine out; however, there is a weight penalty because a 13% overthrust is required. This amount of overthrust is outside the presently designed engine propellant utilization control capability. Uprated or added engines are required with associated weight penalties. Because the FR-4 with the 9-3-9 arrangement does not have fail-operational/fail-safe capability for booster engines, mission aborts are higher than for the FR-3. The FR-4 has fail-safe provisions for engines and basically the same abort procedures as the FR-3 described above. The intact abort success probability (safety) of the FR-4 is therefore approximately the same as for the FR-3.

Both the FR-3 and FR-4 vehicle concepts incorporate inert gas purging provisions for fuel tank surrounds, rocket engine bay, and payload bay to suppress potential fire or explosion resulting from leakage and subsequent vaporization of fuel (LH_2). Purging with an inert gas is provided during ascent and descent to an O_2 concentration < 2% by volume for these areas.

Sealed, gas-tight bulkheads separate compartments containing fuels and/or oxidizers and diaphragms seal off hot air and isolate hot surface ignition sources.

SECTION 2

MISSION REQUIREMENTS

This section presents the NASA mission traffic schedule, a description of each mission, and the performance capabilities of the FR-1 for each mission.

2.1 TRAFFIC MODEL

The Nominal Space Shuttle Traffic Model from the NASA Space Shuttle Task Group Report, Volume 1, 12 June 1969, is the basis for the model presented in Table 2-1. The time period of interest is 1975 through 1985. This model reflects an average annual launch rate of 51 flights per year. Table 2-2 summarizes the range of mission characteristics for the missions shown on Table 2-1. The frequency of mission types is:

Propellant Delivery	44
Personnel and Cargo Delivery	33
Propulsive Stage and Payload Delivery	9
Experiment Module Delivery	6
Satellite Missions	4
Short Duration Orbit Missions Rescue Missions	4
	100 percent

2.2 SPACE STATION/BASE LOGISTICS

The space shuttle transports cargo and personnel to and from a manned orbital space station and subsequently to a larger space base in low-altitude Earth orbit. The cargo includes food, liquids, and gases in addition to both experiment modules and operational equipment. Personnel include trained astronauts, and individuals who conduct specific scientific and technology experiments and operations. The shuttle logistics missions include long-lead-time scheduled resupply and crew rotations as well as discretionary flights. The routine logistics requirements for an orbital facility depend on the size of the facility and the type of experiments and operations being conducted at any given time. Typical requirements are summarized in Table 2-3 for a 12-man space station and a 50-man space base.

Table 2-1. NASA Mission Traffic Model

Mission	Year 75	76	77	78	79	80	81	82	83	84	85	Total Traffic
Logistics												
Space Station												
Personnel and cargo	4	4	4	4	4							20
Experiment module	3	3	3	3	3							15
Space Base												
Personnel and cargo						20	20	20	20	20	20	120
Experiment module						3	3	3	3	3	3	18
Delivery of Propulsive Stages and Payload	7	1	8	3	4	6	5	2	7	5	3	51
Placement, Retrieval, Service, and Maintenance of Satellites	2	2	2	2	2	2	2	2	2	2	2	22
Delivery of Propellant												
LH_2				36	36	24	24	24	24	24	24	216
LO_2				6	6	4	4	4	4	4	4	36
Crew and Cargo (Lunar Mission)				6	6	6	6	6	6	6	6	48
Short Duration Orbit	2	2	2	2	2	2	2	2	2	2	2	22
Space Rescue												
Total Flights Per Year	18	12	19	62	63	67	66	63	68	66	64	568

Table 2-2. NASA Mission Characteristics

Mission	Orbit Altitude (n. mi.)	Orbit Inclination (deg)	On-Orbit ΔV (fps)	Mission Duration (days)	Launch Azimuth (deg)
Space Station Space Base Logistics	200 to 300	28.5 to 90	1000 to 2000	7	90 to 180
Delivery of Propulsive Stage and Payload	100 to 200	28.5 to 55	1000 to 1500	7	90 to 41
Placement, Retrieval, Service, and Maintenance of Satellites	100 to 800	28.5 to Sun Synch	1000 to 5000	7 to 15	90 to 188
Delivery of Propellants	200 to 300	28.5 to 55	1000 to 2000	7	90 to 41
Short Duration Orbit	100 to 300	28.5 to 90	1000 to 2000	7 to 30	90 to 180
Space Rescue	270	55	2000 to 5000	7	41

Table 2-3. Routine Logistics Requirements

Requirements	Space Station (12 men)	Space Base (50 men)
Per Quarter		
Cargo Up, lb	12,000	48,000
Personnel Up, men	12	50
Cargo Down, lb	7,000	28,000
Personnel Down, men	12	50
Per Flight (Based on Traffic Model)		
Cargo Up, lb	12,000	9,600
Personnel Up, men	12	10
Cargo Down, lb	7,000	5,600
Personnel Down, men	12	10

The routine logistics mission is defined as a 55-degree inclined circular orbit at a 270-n. mi. altitude, with rendezvous within 24 hours of launch. The main propulsion system on-orbit ΔV design requirement is 1800 fps and the attitude control system (ACS) ΔV design requirement is 200 fps.

A series of trajectories was prepared to determine the best orbiter burnout altitude. Typical results are shown in Figure 2-1. The payload weight Δ includes the effects of a Hohmann transfer to a 100-n. mi. orbit. Typical ΔV losses for a staging velocity of 8400 fps and a staging dynamic pressure of 50 psf are:

Burnout Altitude, n. mi.:	50	43
Gravity, fps	520	412
Drag, fps	19	35
Misalignment, fps	357	351
Total	896 fps	798 fps

As shown, there is a decrease of about 100 fps in gravity losses for the lower injection altitude. This more than offsets the higher drag losses and the additional ΔV to transfer from the lower altitude. The effect of staging dynamic pressure (staging altitude) is small.

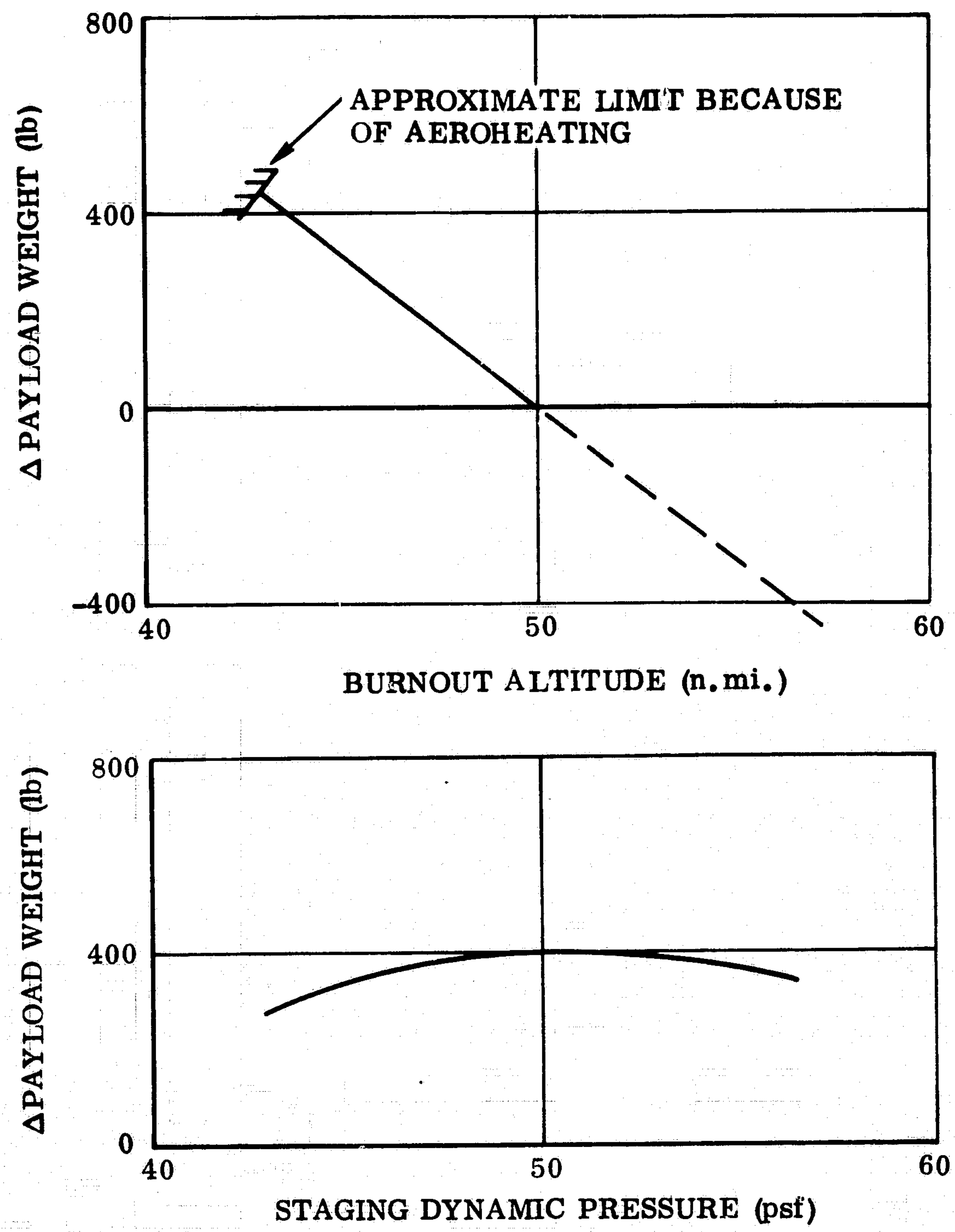


Figure 2-1. Δ Payload vs Burnout Altitude and Staging Dynamic Pressure

Where possible, launches that require rendezvous with a space station or satellites will be made in-plane and in-phase. Where this is not possible, the launch will be made in-plane but out of phase. Table 2-4 shows candidate parking orbits and their characteristics.

Table 2-4. Candidate Parking Orbits

	Parking Orbit (n. mi.)			
	Circular			Eliptical
	43	80	100	43 by 270
Time for 360-degree Phasing with 270 n. mi.* hr	16.9	20.6	23.1	32
Drag, lb	14,000 (const)	4	0.9	14,000 (peak)
Δh Loss/Day, n. mi.		7	1.7	
ΔV Loss/Day, fps	110,000 +	25	6.2	4,000
Heating	bad	good	good	bad
No of Engine Starts Required	2 +	2	2	> 1, ≤ 21

*All inject at 43 n. mi. altitude. Times include Hohmann transfer to appropriate parking orbit plus Hohmann transfer to 270 n. mi.

The lowest parking orbit (43 n. mi. circular) has the least waiting time (16.9 hours) to change phase of 360 degrees with a space station at 270 n. mi. However, this low altitude is impractical because of high drag and heating. The 43 by 270 n. mi. elliptical parking orbit is also a poor choice because of perigee drag and heating and the time required to match phase. The 80 and 100-n. mi. circular orbits are both reasonable, with the lower altitude slightly better because of reduced phasing time but worse in terms of drag. Because of drag, parking orbits below 80 n. mi. are not very practical if phasing angles up to 360 degrees are required. A 100-n. mi. parking orbit was selected for the baseline mission since it meets the 24-hr rendezvous requirement and requires less ΔV for drag makeup.

Figure 2-2 presents the mission profile showing main engine burns. The main propulsion system ΔV requirements are shown in Table 2-5. The 1800 fps requirement shown contains an allowance of 200 fps for insertion dispersions and out-of-plane errors, and 480 fps for flight performance reserve (FPR) and contingencies.

The ACS furnishes limit cycle attitude control to +45 degrees while in orbit hold or during orbit transfers, orientation to ± 5 degrees prior to each orbit maneuver burn and during rendezvous, roll control to ±0.5 degrees during each maneuver burn, ΔV to transfer from 260 n. mi. to 270 n. mi., rendezvous, dock, and undock, and orientation

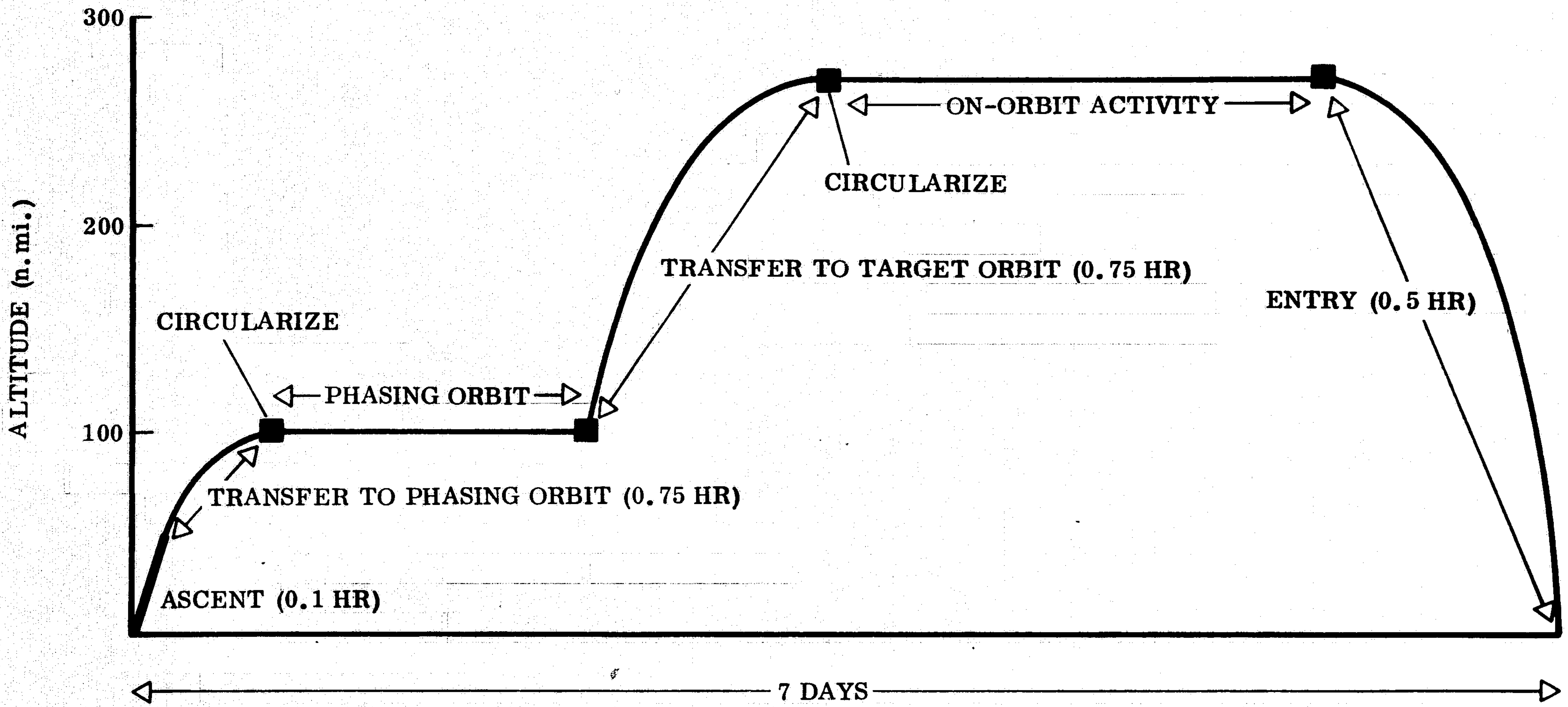


Figure 2-2. Logistics Mission Profile

Table 2-5. Main Propulsion ΔV Requirements

Maneuver	ΔV (fps)
Circularize at 100 n. mi.	100
Transfer to 260 n. mi.	280
Circularize at 260 n. mi.	280
Entry	450
FPR and Contingencies	480
Insertion Dispersions and Out-of-Plane Errors	200
Total System ΔV	1800

to ± 2 degrees during entry. The total ACS ΔV for the mission is 157.9 fps. The ΔV requirements for each mission phase are shown on Table 2-6. The design requirement of 200 fps indicates a reserve and dispersion allowance of 42.1 fps.

2.3 ALTERNATE MISSION PERFORMANCE ANALYSIS GROUND RULES

The alternate missions, shown in Table 2-2 below the baseline mission, must be assessed to determine their performance requirements relative to the baseline mission. The discretionary payload for these alternate missions is composed of propulsion capability for vehicle maneuvers above the baseline mission requirements, mission support equipment, expendable and reusable propulsion stages for payload transfer to high-altitude orbits, and the payload itself.

An analysis to determine the sensitivity of FR-1 performance to changes in payload weight and total propellant loaded was conducted. The analysis was based on ascent trajectory and performance data of appropriate synthesis runs with the following ground rules:

Perigee Altitude	260,000 ft (43 n. mi.)
Perigee Inertial Velocity	25,897 fps
Orbit	55° Inclination (ETR)
Maneuver ΔV (Main Propulsion System)	1800 fps

Figure 2-3 shows the approximate decrease in ideal velocity at perigee injection as a function of payload weight increase computed from the impulsive logarithmic velocity formula:

Table 2-6. ACS ΔV Requirements

Task	Function	ΔV (fps)	Task	Function	ΔV (fps)
Limit cycle during transfer and orbit coast for 20 hours	Attitude Control to ± 45 deg	9.7	Rendezvous and dock	Translate	7.2
	Drag makeup	5.2		Attitude Control to ± 5 deg for 15 minutes	1.1
Orbit Maneuvers	Orientation to ± 5 deg for 20 minutes prior to maneuver. Four maneuvers	5.9		Attitude Control to ±0.5 deg for 20 seconds	1.7
	Control roll during maneuvers. ± 5 deg	1.0	Undock	Translate	8.3
Transfer from 260 to 270 n. mi.	Transfer ΔV	16.9		Attitude Control to ±0.5 deg for 20 seconds	1.7
	Attitude Control to ± 0.5 deg for 22 seconds before and during burn	3.1	Limit cycle for 24 hours	Attitude Control to ±45 deg	11.6
	Attitude Control to ± 5 deg during 0.75 hour transfer	3.3	Entry	Control to ±2 deg with 2.5 deg/sec^2	23.1
Circularize at 270 n. mi.	Transfer ΔV	16.9		Control to ±2 deg with 1.9-2.25 deg/sec^2	26.0
	Attitude Control to ±0.5 deg for 22 seconds before and during burn	3.1		Control to ± 2 deg with 0.75-1.25 deg/sec^2 for 750 sec	11.5
				Total ACS ΔV Requirement	157.9

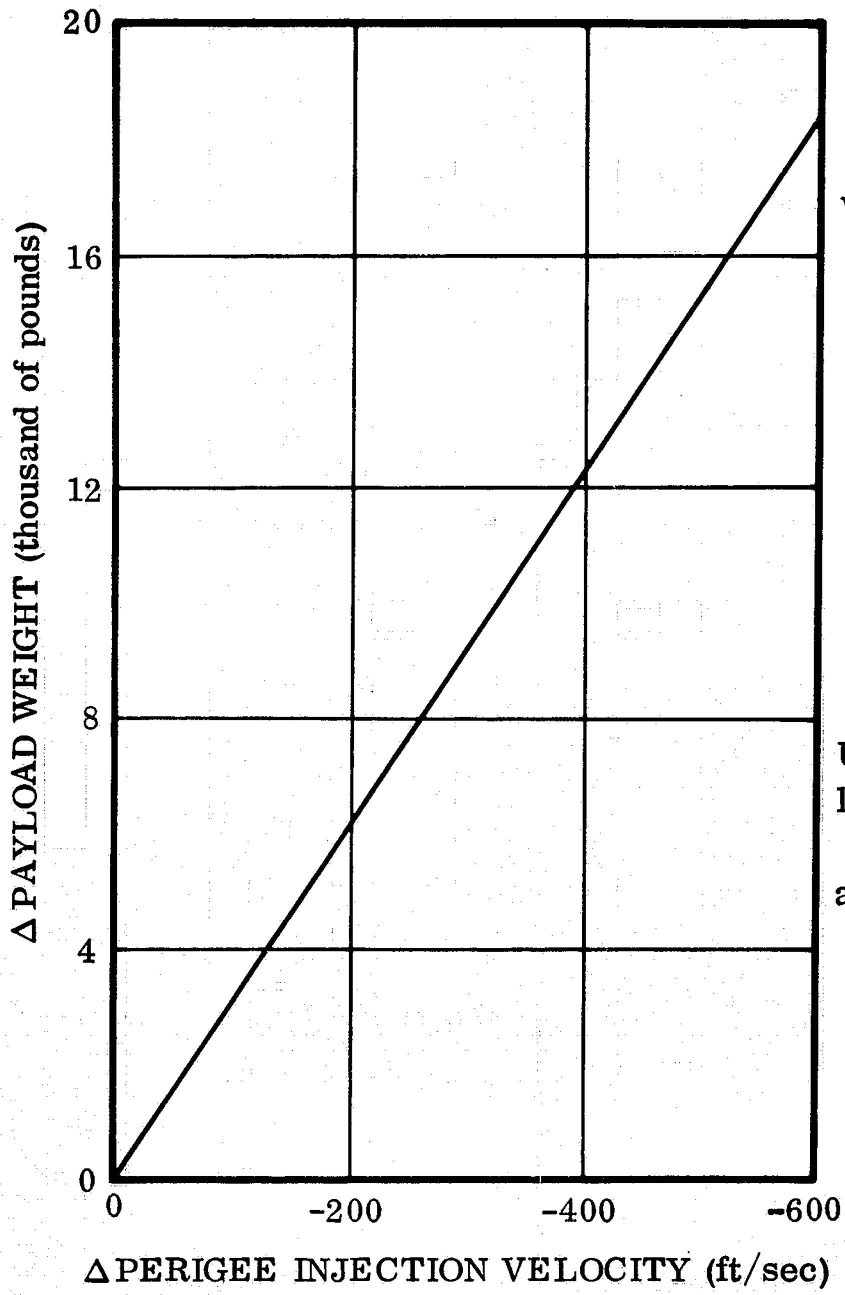


Figure 2-3. Injection Velocity Loss Due to Payload Increase

$$\Delta(\Delta V) = g_o \left[I_1 \, \Delta(\ln \mu_1) + I_2 \, \Delta(\ln \mu_2) \right]$$

where

g_o = 32.174 ft/sec^2

I_1 = Booster vacuum I_{sp}

I_2 = Orbiter vacuum I_{sp}

μ_1 = Booster mass ratio

μ_2 = Orbiter mass ratio

Use of this curve is subject to the following qualifications.

a. Use of the ideal velocity formula assumes constant velocity losses. The actual increase in velocity losses resulting from the lower thrust-to-weight values (higher payload values) is about one fps per thousand pounds of payload. Thus the slope of the curve should be decreased by three percent.

b. The increase in payload above 50,000 pounds was derived by analysis of the ascent trajectory only. The FR-1 configuration is not designed to enter and land with the heavier payloads. Abort constraints may also require that heavier payloads be partially jettisonable.

Figure 2-4 presents the change in orbital maneuver velocity as a function of payload weight and additional propellant tanked into the payload bay. The payload bay was taken as a cylinder, 15 feet in diameter and 60 feet long. If the payload does not occupy the entire bay, it was assumed that a cylindrical propellant tank could be placed in the remaining volume. In performing this portion of the analysis, the following assumptions were made:

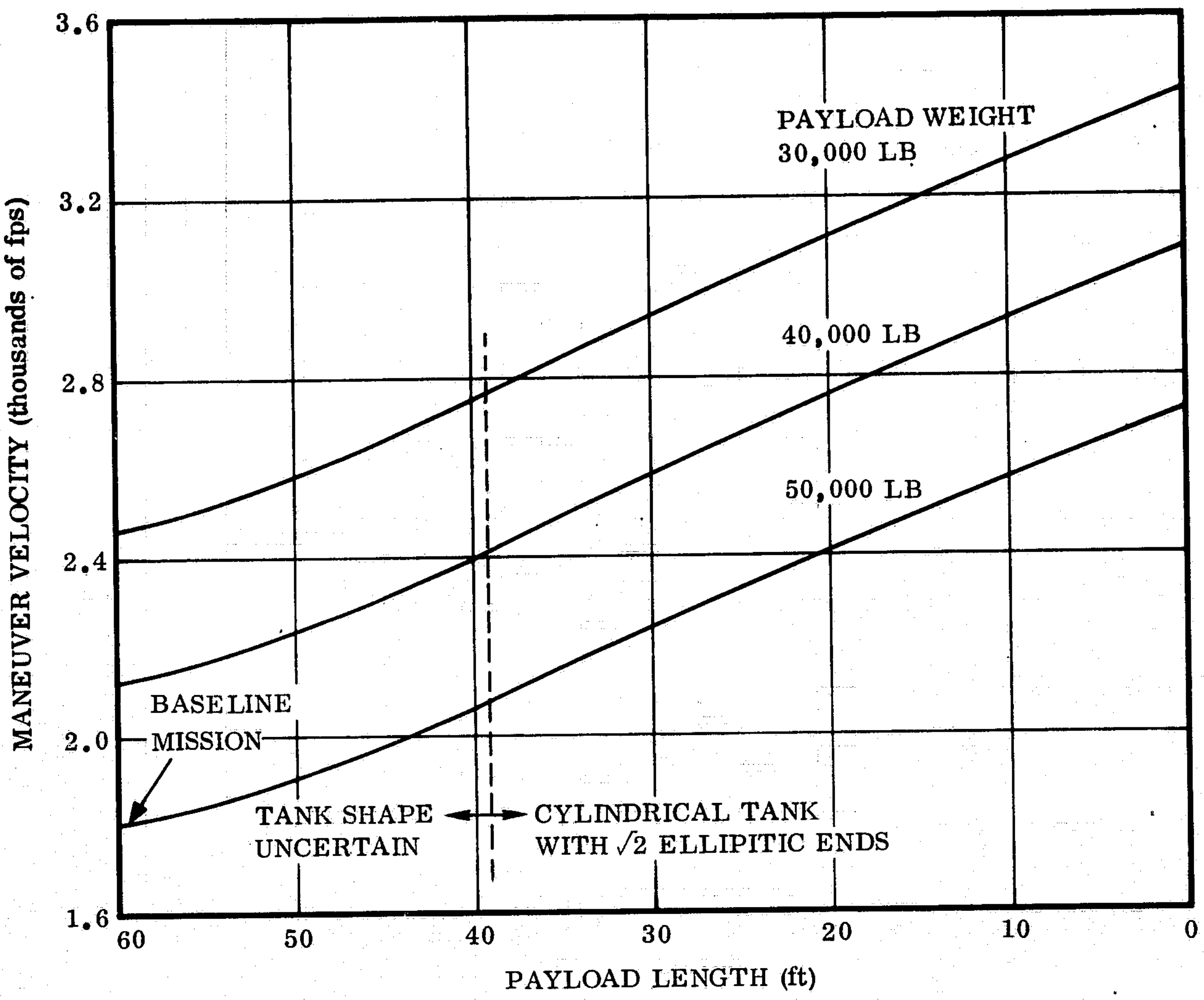


Figure 2-4. Maneuver Velocity as a Function of Payload Weight and Propellant Loading in the Payload Bay

Propellant mixture ratio = 7:1

Maneuver I_{sp} = 427 sec

Hydrogen density = 4.2 lb/ft^3

Oxygen density = 71.0 lb/ft^3

0.05 lb of structural and tankage weight added per lb of added propellant

Two inches of insulation covering outside of propellant tank (included in structural weight)

Single propellant tank with internal separating bulkhead and $\sqrt{2}$ elliptic ends.

The last assumption dictates that the smallest possible tank is 20.7 feet long and is the limiting case of two elliptic ends. Tanks smaller than this would have to take annother (undefined) shape. Since the amount of propellant per foot of payload bay length is unknown for this condition, the curves shown on Figure 2-4 are uncertain for payload length values greater than 38.9 feet.

The maneuver velocity analysis was made with a series of trajectory simulations. The amount of propellant remaining for orbital maneuvering was calculated and translated into maneuver velocity using the impulsive velocity formula. Booster staging for all runs was held constant at a q of 50 psf. A linear partial of 15 fps per foot of propellant tank length was used for payload estimates.

The inertial velocity due to launch azimuth is shown on Figure 2-5 for ETR and WTR. The accuracy of the simple velocity addition used is compatible with the accuracy of the alternate mission on-orbit ΔV requirements. The data are also applicable for launches from 180°to 360° using a negative velocity component.

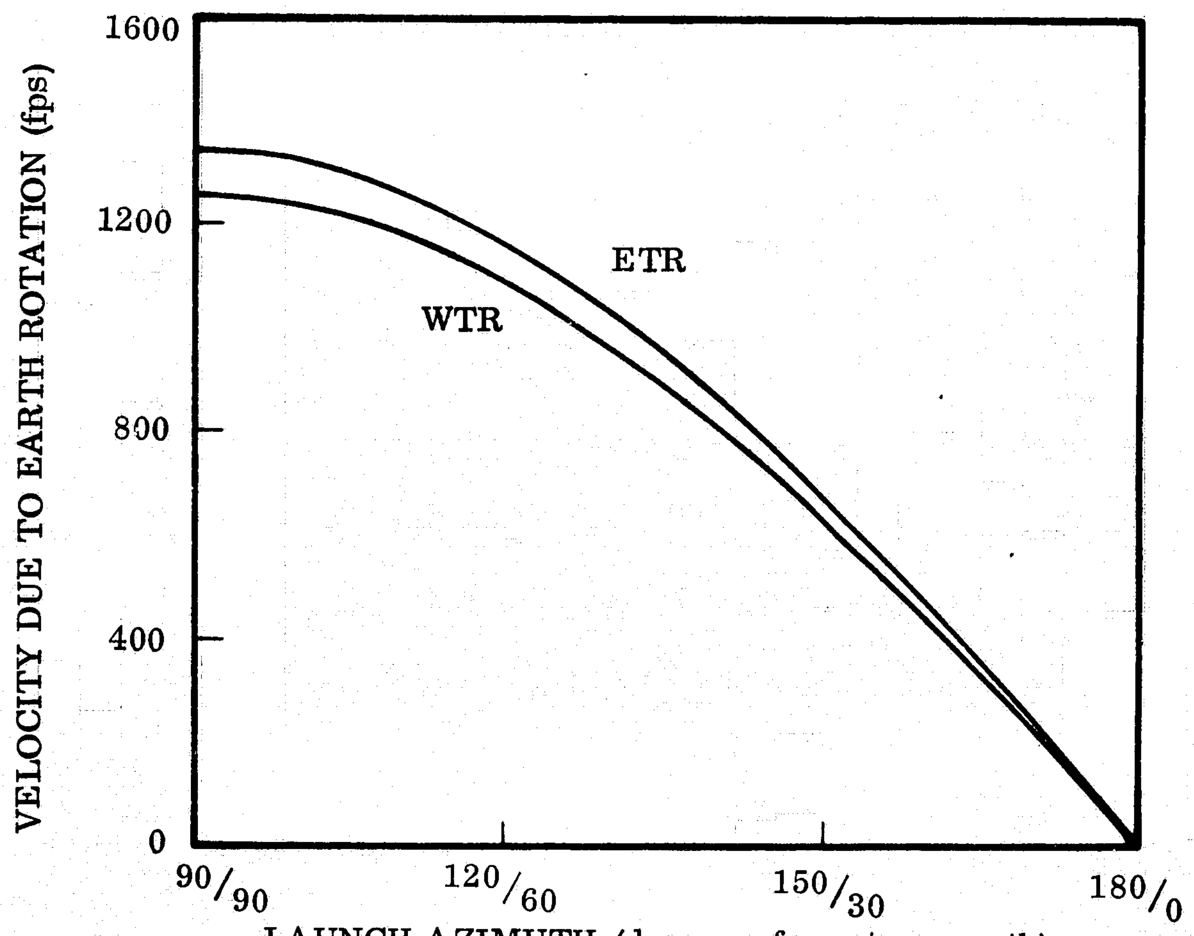


Figure 2-5. Velocity Increment Due to Launch Azimuth

All azimuth launch capability involves consideration of range safety, sonic boom, and rocket engine noise. Range safety for expendable launch vehicles is normally concerned with impact of the launch vehicle during normal flights and impact of portions of the system due to failures.

Launch vehicle impact, however, is eliminated since the booster portion of the system is recovered by flyback to the launch site. No objects are jettisoned or released during the flight.

Vehicle subsystem failures should not result in vehicle or mission loss due to redundancy incorporated in the basic design. Analysis of the critical subsystem for mission

loss (the rocket engines) shows a fail operational, fail safe design is possible permitting mission completion with one booster engine out, and vehicle recovery without mission completion for two booster engines out. With this concept it is possible to keep vehicle losses within the range of 0.1 to 1.0 per 1000 flights. The major parameter determining the exact value is single engine reliability. It is assumed for this discussion on all azimuth launch capability that the vehicle loss rate is low enough to relax normal range safety constraints.

A typical boost trajectory was used to examine the sonic boom problem. As the boost configuration reaches Mach 1 the flight path is close enough to vertical that the shock wave propagates away from the ground and is thus not heard at ground level. As the boost continues and the flight path is depressed, a point is reached some three miles downrange where the shock wave does intersect ground level, and a very mild boom is felt (overpressure of only 0.7 psf). At a downrange distance of some six miles, the sonic boom is so weak that it is not heard at all at ground level.

The noise generated by rocket engines during booster phase of the space shuttle is of concern because of the possible detrimental effects on the vehicle itself and persons near the launch site. Data from engine firings and expendable vehicle launches in conjunction with a typical trajectory profile were used to generate the noise level as a function of vehicle range. This relationship is shown on Figure 2-6. Noise at liftoff of approximately 180 db reduces to the threshold of pain (140 db) at a range of 1400 feet and the threshold of discomfort (120 db) at 13,000 feet. Typical safety requirements require at least 9500 feet from the launch pad to inhabited buildings due to the explosive hazard of the propellant weight in the space shuttle. The noise level at 9500 feet is 123.5 db and is assumed to be acceptable. The buildings at this range would be only those associated with the space shuttle operation.

No apparent problems other than range safety release were uncovered in the analysis. The all-azimuth launch capability should be considered in future analyses of vehicle performance.

2.4 DELIVERY OF PROPULSIVE STAGES AND PAYLOAD

The space shuttle delivers propulsion stages and payloads to low Earth orbits to support a variety of missions within Earth orbit and out of Earth orbit. Such missions range from high altitude Earth satellites to unmanned planetary probes. In this operational mode, the space shuttle delivers both the payload package and the propulsive stage to orbit in a single launch. Upon achieving a low Earth orbit (100 to 200 n.mi. circular), the propulsive stage and payload are checked out and launched by the special two-man launch team carried on the orbiter.

On-orbit staytimes of up to seven days are required to allow for on-orbit checkout and launch window phasing. The two predominant propulsive stages to be used are Centaur and AJ-10-138. The Centaur characteristics are shown in Figure 2-7.

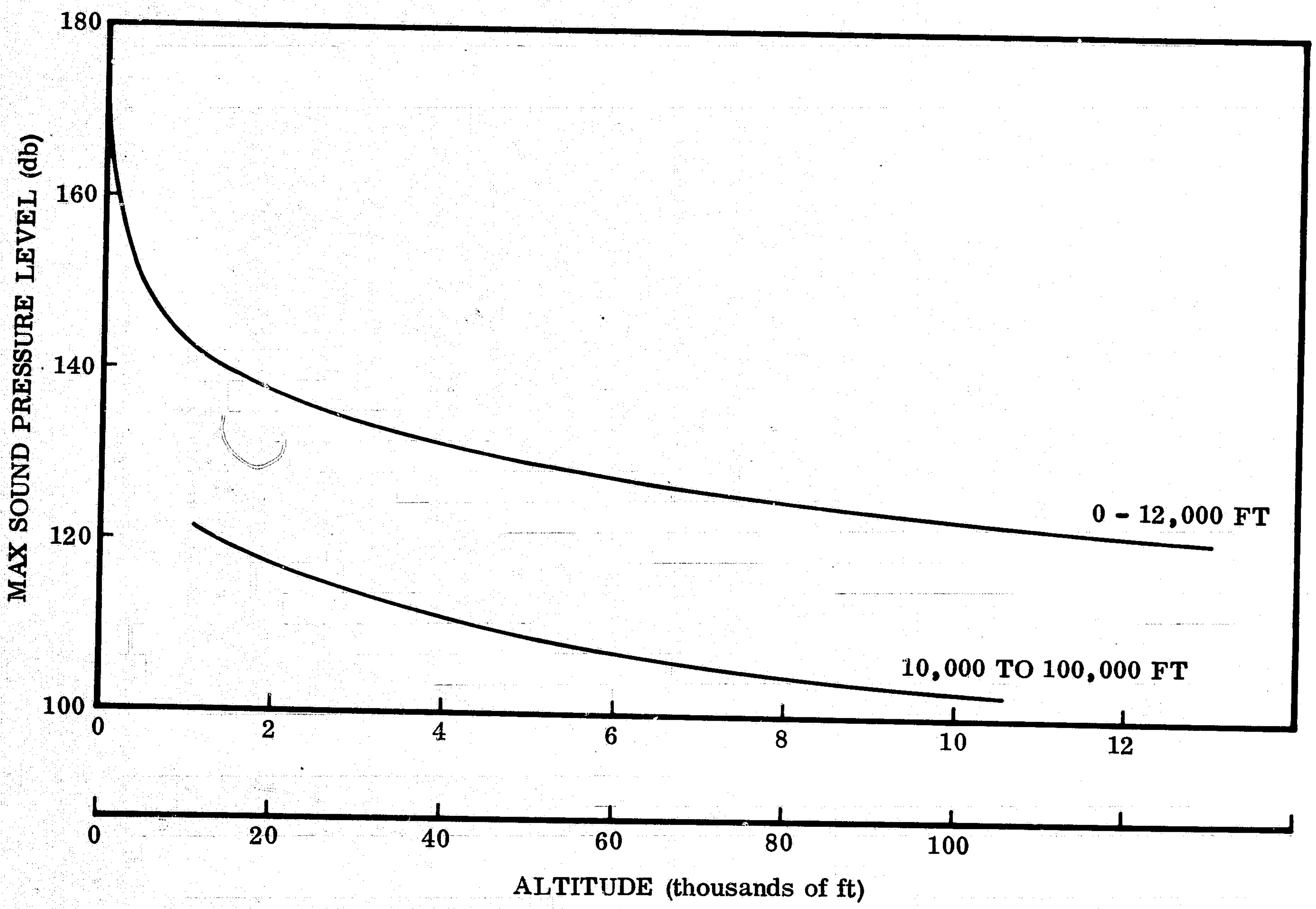


Figure 2-6. Noise Level of Space Shuttle During Booster Phase

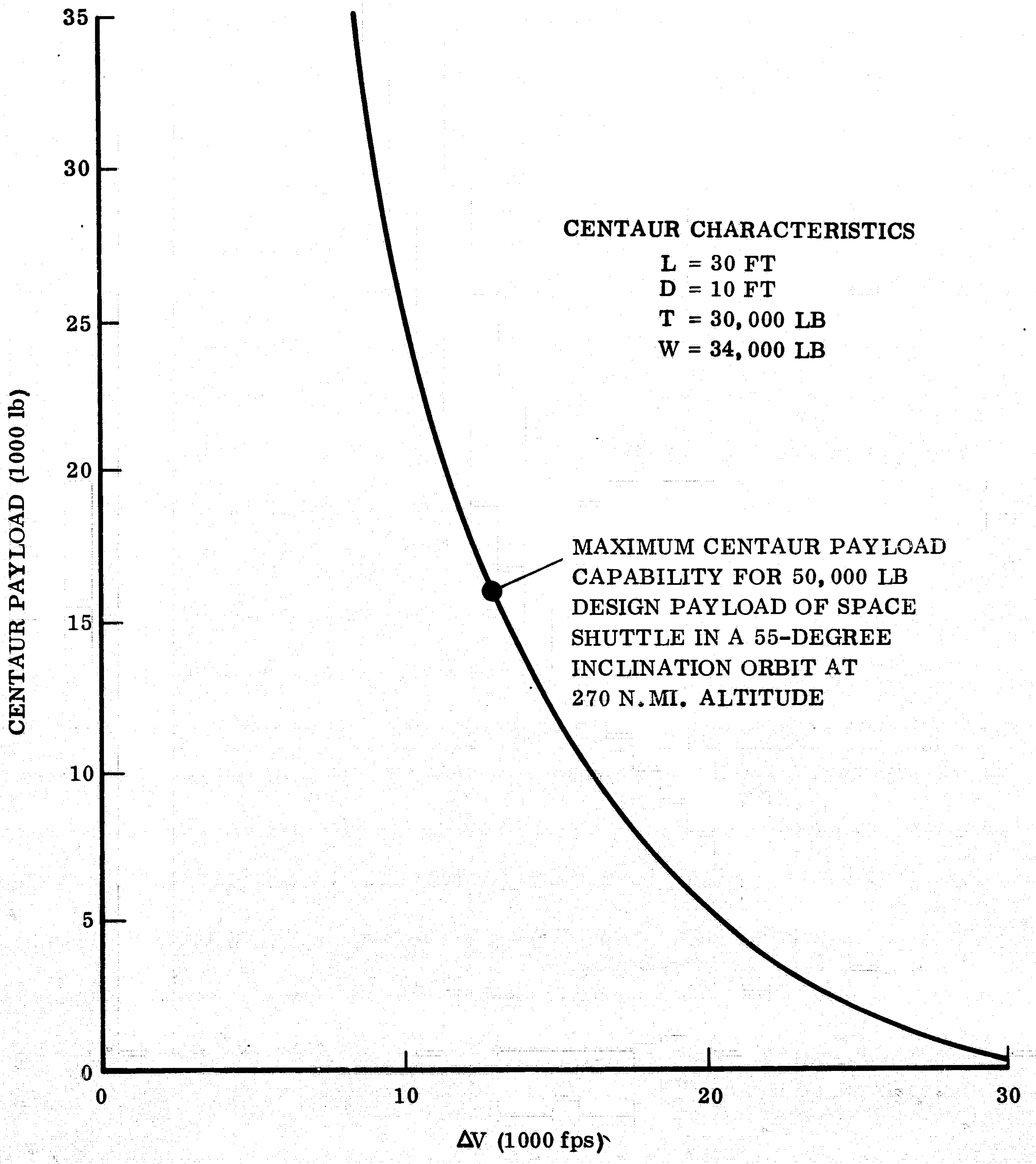


Figure 2-7. Centaur Payload Capability for Low Earth Orbit Launch

The ΔV requirements for the highest payload capability orbit (100 n. mi. at 28.5 degrees orbital inclination) and the lowest payload capability orbit (200 n. mi. at 55 degrees) are shown in Table 2-7 as a function of mission profile. No parking orbit is assumed for phasing. The resulting payload capability range based on a linear partial of the orbiter of -30.8 pounds of payload per fps is 65,400 to 88,600 pounds. The latter is for the due-east launch (inclination = 28.5 degrees) from ETR into a 100 n. mi. orbit.

Table 2-7. Delivery of Propulsive Stages and Payload ΔV Requirements

Orbital Altitude (n. mi.)	100		200	
Orbital Inclination (deg)	28.5		55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n. mi.	110	—	—	—
Transfer to 200 n. mi.	—	—	160	—
Circularize at 200 n. mi.	—	—	180	—
Drag Makeup	—	45	—	—
Undock	—	10	—	10
Entry	300	50	380	50
Dispersion	200	20	200	20
FPR and Contingency	400	—	400	—
Total ΔV	1010	125	1300	80
ΔV Difference from Baseline	-790	-75	-500	-120
ΔV Difference due to Launch Azimuth change from Baseline	-460	—	0	—

2.5 PLACEMENT, RETRIEVAL, REPAIR, AND MAINTENANCE OF SATELLITES

The space shuttle can place unmanned satellites into various Earth orbits. It can also revisit certain high-priority or high-cost satellites and return them to Earth if necessary. For such missions, the shuttle will be required to operate at altitudes up to 800 n. mi. and orbit inclinations from 28.5 degrees to polar. With this versatile operational capability, a wide variety of unmanned satellites will be prime candidates for space shuttle support.

These satellites are also logical candidates to be serviced and maintained by the space shuttle. The orbiter would then require the capability to revisit modules and satellites and bring them into an onboard facility where a service and maintenance crew could

work in a shirtsleeve environment. The orbiter service and maintenance facility would contain equipment, instruments, and supplies that would allow trained personnel to conduct:

a. Routine Servicing and Maintenance. These periodic functions would include such items as film changing and replenishment of attitude control propellants.

b. Repair. Although highly automated satellites are designed for long-term operations, a capability to visit such satellites in case of malfunctions is highly desirable. The oribiter could provide for on-orbit replacement of instruments and components.

The payload capability range for the satellite placement mission is based on the ΔV requirements in Table 2-8. The two orbits defining the range are 100 n. mi. at 28.5 degrees inclination and 800 n. mi. at 90 degrees inclination. The satellite repair or retrieval missions will require more ΔV due to rendezuous requirements (within 24 hours of launch), as indicated in Table 2-9 for the same two trajectories. The payload range is 88,600 to 0 pounds for placement and 56,800 to 0 pounds for service or retrieval.

Table 2-8. Satellite Placement ΔV Requirements

Orbit Altitudes (n. mi.)	100		800	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n. mi.	110	—	—	—
Transfer to 800 n. mi.	—	—	1105	—
Circularize at 800 n. mi.	—	—	1080	—
Drag Makeup	—	45	—	—
Undock	—	10	—	10
Entry	300	50	1225	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	1010	125	4010	90
ΔV Difference from Baseline	-790	-75	+2210	-110
ΔV Difference from Baseline due to Launch Azimuth	-460	—	+880	—

Table 2-9. Satellite Service or Retrieve Mission ΔV Requirements

Orbital Altitude (n. mi.)	100		800	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Transfer to 250 n. mi.	260	—	—	—
Circularize at 250 n. mi.	360	—	—	—
Transfer to 100 n. mi.	260	—	—	—
Circularize at 100 n. mi.	260	—	—	—
Circularize at 100 n. mi.	—	—	110	—
Transfer to 800 n. mi.	—	—	1130	—
Circularize at 800 n. mi.	—	—	1080	—
Terminal Phase	—	20	—	20
Braking/Stationkeeping	—	90	—	90
Docking	—	10	—	10
Drag Makeup	—	90	—	—
Undocking	—	10	—	10
Entry	300	50	1225	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	2040	290	4145	200
ΔV Difference from Baseline	+240	+90	+2345	0
ΔV Difference due to Launch Azimuth	-460	—	+880	—

2.6 DELIVERY OF PROPELLANT

The space shuttle would operate as a propellant-delivery tanker in conjunction with a long-duration orbital propellant storage (OPS) facility. The OPS facility would act as a filling station to supply liquid hydrogen and oxygen propellants for high-energy, large-payload propulsive stages for interplanetary missions which could not be launched from Earth fully loaded with the space shuttle, for space-based vehicles operative between Earth orbit and the moon, and within Earth orbit. Propellants will also be delivered to the spacebase/station.

When operating as a propellant tanker, the orbiter payload bay is configured differently depending on whether it is delivering all liquid hydrogen, liquid oxygen and liquid hydrogen, or a mix of dry cargo and propellants. The largest payload volume requirement will be for liquid hydrogen deliveries. Including the tankage, insulation, and propellant transfer mechanisms, 45,000 pounds of liquid hydrogen would require about a 50,000-pound capability for the space shuttle. The volume corresponding to 45,000 pounds of liquid hydrogen is about 11,000 ft^3. With the volume required for liquid hydrogen, sufficient capability will exist for combined liquid hydrogen and oxygen loads.

The space shuttle will rendezvous with the OPS facility and transfer propellant without crew EVA. Two men in addition to the crew will monitor the operation and provide manual override to the transfer systems.

The lunar mission supply is composed of six men and 20,000 pounds of payload to be delivered to the OPS where the lunar tugs will be located, serviced, and fueled. Other payloads for interplanetary missions will also be delivered to the OPS for integration with the space tug or nuclear shuttle.

The mission duration is seven days. Table 2-10 presents the ΔV requirements for the highest payload mission (200 n. mi. at 28.5 degrees inclination) and the lowest payload mission (300 n. mi. at 55 degrees). Rendezvous is within 24 hours of launch. The resulting payload range is 74,900 to 46,600 pounds.

2.7 SHORT-DURATION ORBIT

The space shuttle will be capable of operation as a short-duration orbital station for up to 30 days to exploit man's capabilities as a selective sensor and decision maker.

The higher resolution obtained from a 100-n. mi. orbit as opposed to a 270-n. mi. orbit indicates a unique capability of the space shuttle short-duration orbital mission even in a 55-degree (or lower) orbital inclination. The large payload volume capability of the orbiter provides an ideal platform for the development of advanced equipment and instrumentation.

A module or modules are required containing appropriate instrumentation and provisions for a 10-man crew in a shirtsleeve environment. This module can also be used as a flying test bed for sensor research, development, test, and calibration to support both manned and unmanned satellite missions, to develop and test complete experiment systems to verify their operational capabilities before being integrated into the space station, and to develop and flight test systems components in support of a manned planetary program.

The payload capability ranges from 88,600 pounds for a 100 n. mi. orbit at 28.5 degrees inclination to 22,300 pounds for a 300-n. mi. orbit at 90 degrees. The ΔV requirements associated with these missions are presented in Table 2-11.

Table 2-10. Delivery of Propellant Mission ΔV Requirements

Orbit Altitude (n. mi.)	200		300	
Orbit Inclination (deg)	28.5		55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n. mi.	110	—	110	—
Transfer to 200 n. mi.	180	—	—	—
Circularize at 200 n. mi.	180	—	—	—
Transfer to 300 n. mi.	—	—	350	—
Circularize at 300 n. mi.	—	—	350	—
Transfer Phase	—	20	—	20
Drag Makeup	—	—	—	—
Dock	—	10	—	10
Undock	—	10	—	10
Entry	380	50	500	50
FPR and Contingency	400	—	400	—
Dispersions	200	20	200	20
Total ΔV	1450	110	1910	110
ΔV Difference from Baseline	-350	-90	+110	-90
ΔV Difference from Baseline Launch Azimuth Change	-460	—	—	—

2.8 RESCUE

The space shuttle capability for space base/station rescue requires rendezvous within 24 hours of the rescue request. The ΔV requirements in Table 2-12 reflect worst-case phasing requiring a 16-hour wait for the launch window, with 8 hours remaining for the flight operation to arrive at the base. These ΔV requirements result in no payload capability.

An increase in the allowable time to rendezvous from launch, or a better space base location at the time of rescue request will result in improved payload capability. The use of a main engine propellant tank in a portion of the payload bay will also increase payload capability as discussed in Section 2.3.

Table 2-11. Short-Duration Orbit ΔV Requirements

Orbit (n. mi.)	100		300	
Orbit Inclination (deg)	28.5		90	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Circularize at 100 n. mi.	110	—	—	—
Transfer to 300 n. mi.	—	—	370	—
Circularize at 300 n. mi.	—	—	350	—
Drag Makeup	—	190	—	—
Station Keeping	—	360	—	360
Entry	300	50	500	50
FPR and Contingencies	400	—	400	—
Dispersion	200	20	200	20
Total ΔV	1010	620	1820	430
ΔV Difference from Baseline	-790	+420	+20	+230
ΔV Difference due to Launch Azimuth Change	-460	—	+880	—

Table 2-12. Rescue Mission ΔV Requirements

Orbit Altitude (n. mi.)	270	
Orbit Inclination (deg)	55	
	Main Propulsion ΔV (fps)	ACS ΔV (fps)
Transfer to 550 n. mi.	750	—
Circularize 550 n. mi.	720	—
Transfer to 270 n. mi.	720	—
Circularize 270 n. mi.	750	—
Transfer Phase	—	20
Braking/Station Keeping	—	90
Dock	—	10
Undock	—	10
Entry	450	50
FPR and Contingency	400	—
Dispersion	200	20
Total ΔV	3990	200
ΔV Difference from Baseline	+2190	—

SECTION 3

FLIGHT OPERATIONS

This section presents the launch to landing operations with emphasis on rendezvous and docking. The ascent, entry, and landing phases are straightforward operations. The major tradeoffs appear in the rendezvous, docking, and on-orbit payload handling phases. A summary of the on-orbit payload operation analysis is included here; this flight phase is covered in further detail in Section 4.

3.1 ASCENT

The events during this phase are almost identical with any expendable type system up to separation. For the space shuttle, of course, the booster also performs entry, maneuver, and landing. The events are as follows:

a. Liftoff.

b. Monitor vehicle systems during ascent.

c. Perform roll program to desired azimuth.

d. Perform pitchover program.

e. Perform separation maneuver.

f. Verify separation —
   Determine trajectory for booster.
   Start attitude program for booster entry.
   Start booster entry.

g. Start orbiter engines.

h. Monitor orbiter thrust vector control and ascent trajectory.

i. Monitor environmental control system.

j. Initiate orbiter engine cutoff sequence.

k. Confirm safe Earth orbit.

l. Activate attitude control as required for coast to 100-n. mi. apogee.

m. Maneuver vehicle in firing attitude.

n. Determine and activate thrust program for insertion into parking orbit.

o. Verify time required to achieve correct phasing.

p. Monitor onboard guidance and navigation data.

q. Maneuver vehicle in firing attitude and settle propellants.

r. Determine and activate thrust program for ascent to target.

s. Acquire target during coast.

t. Determine thrust program to achieve gross rendezvous.

3.2 RENDEZVOUS AND DOCK

Rendezvous and docking requirements vary significantly from mission to mission, and in some cases within missions, as illustrated in Table 3-1.

Table 3-1. Rendezvous and Docking Requirements

Mission	Rendezvous Requirement	Docking Requirement	Cooperative Target
Space Station Logistics	Yes	+	Yes
Satellite Placement	No	No	Not Appl.
Satellite Retrieval	Yes	Yes	+
Delivery of Stages and Propellants	Yes	Subject of Tradeoff	Yes
Satellite Maintenance	Yes	Subject of Tradeoff	+
Short Duration Orbital	No	No	Not Appl.
Rescue	Yes	+	+

+Undetermined

The requirement for rendezvous is implicit in each type of mission. The requirements for docking, however, are not clear. For example, personnel and cargo transfer concepts currently under study include both docking and non-docking techniques. The same is true for satellite maintenance concepts, as well as for experiment module transfer methods to the space station/base.

For most missions, a cooperative target can be assumed, with corner reflectors as a minimum. Some target failure modes may preclude the use of active cooperative devices on rescue missions or satellite maintenance/retrieval missions. Retrieval might additionally impose a requirement for rendezvous with an uncooperative target.

In performing the rendezvous and docking functions operational options include alternate methods of orbit transfer, manual vs. automatic techniques, active vs. passive targets, and highly accurate navigation vs. long range target acquisition. These options are discussed in the following sections.

3.2.1 COELLIPTIC vs ORBIT-TO-ORBIT TRANSFER. In the U.S. rendezvous programs to date the coelliptic approach has been employed. This is characterized by a plane change (when necessary) to a concentric orbit, phasing maneuvers at either a higher or lower altitude than that of the target, a slow catch-up and final braking maneuver. The advantages of coelliptic transfer include the experience gained on past programs, lower risk due to relatively low ΔV per maneuver, and opportunity for rendezvous at least once per orbit. The primary disadvantages are higher propellant consumption and longer time to rendezvous.

Orbit-to-orbit transfer is accomplished with only two maneuvers: the first to achieve target intercept and the second a braking maneuver. Advantages include shorter time to rendezvous and lower propellant expenditures. Disadvantages include higher risk due to high ΔV per maneuver, fewer opportunities for rendezvous (one in some cases) and a requirement for extremely accurate navigation and guidance.

In view of the high risk associated with orbit-to-orbit transfer, it appears desirable to initially rely on the coelliptic approach. The question then is whether to design for an eventual orbit-to-orbit capability, or to retrofit after confidence in orbit-to-orbit transfer has been developed on some other program.

3.2.2 MANUAL vs AUTOMATIC TECHNIQUES. The use of automatic techniques for rendezvous and docking is attractive from the standpoint of propellant expenditure efficiency, less stringent astronaut training requirements, and suitability for either coelliptic or orbit-to-orbit transfer. Manual techniques, although less efficient and not well suited for orbit-to-orbit transfer, afford greater flexibility for a wide variety of targets, as well as using man's capability to react in off-nominal situations. The most efficient, flexible, and reliable approach is to provide both automatic and manual capability for the space shuttle. This would permit development of automatic techniques at low risk.

Assuming the dual approach, further options are available, ranging from completely independent manual and automatic systems to a completely integrated man/machine system. Sensor tradeoffs include radars vs. lasers for an automatic system, direct vision (windows) vs. indirect vision (mirrors) vs. electro-optical techniques (TV, IR) for a manual system, or some combination of sensors for an integrated system. These tradeoffs are discussed in more detail in Volume 7 of this report.

3.2.3 ACTIVE vs PASSIVE TARGETS. Design requirements for the orbiter will be less stringent if it is assumed that rendezvous targets will be equipped with active devices in the form of radars, lasers, optical beacons or RF beacons. This will not

always be the case, however. Objects presently in orbit which may be targets for retrieval, as well as where failures preclude the use of active devices, must be accommodated. In addition, rendezvous with an actively cooperative target involves more operational complexity, which is to be avoided.

For these reasons, as well as from the standpoint of multi-program costs, the use of passive target techniques is recommended. Rendezvous targets can be readily equipped with corner reflectors, significantly reducing aperture and/or power requirements for the orbiter.

To accommodate retrieval rendezvous with targets now in orbit, provision of additional capability as part of the payload would reduce the requirement on the orbiter. Improved tracking accuracy and ephemeris data potentially available in the mid-70's coupled with orbiter navigation accuracy, may simplify acquisition of non-cooperative targets.

3.2.4 HIGH ACCURACY NAVIGATION vs LONG-RANGE ACQUISITION. The tradeoff between highly accurate navigation and long-range search capability involves the obvious considerations of cost, weight, power, and reliability, together with the following items:

a. Target ephemoris. Regardless of orbiter navigation accuracy, the search capability must accommodate target ephemeris uncertainties.

b. Navigation accuracy. A solution sub-optimized within the navigation and guidance subsystem may very well result in less than optimum system performance.

c. Orbit transfer. Selection of orbit-to-orbit transfer techniques, or a decision to provide for future development of this capability, will impose more severe constraints on navigation and guidance accuracy than the coelliptic method.

d. Other mission requirements. Satellite placement missions and/or short duration orbital missions may require higher navigation accuracy than would be indicated on the basis of the rendezvous requirement alone.

e. Ground tracking and mission control. This is the subject of a major tradeoff, involving several other U.S. space programs.

3.2.5 DOCKING vs NON-DOCKING TECHNIQUES. During the initial stages of the Space Station Program, the orbiter will be considerably larger than the space station. One approach is to dock the station to the orbiter, rather than the orbiter to the station. After docking, the question of attitude control remains. Should the station have the capability for control of the combined mass of the station and orbiter, or should the attitude control function be handed over to the orbiter while docked?

Another alternative is to eliminate docking of the orbiter with the space station. Similar tradeoffs exist for missions other than space station logistics. A number of concepts have been investigated for payload handling with and without docking. These concepts, discussed in detail in Section 4, are summarized in Section 3.3.

3.2.6 DOCKING APPROACH CORRIDORS. Restrictions will be imposed by space traffic in the vicinity of the space station/base; by nuclear radiation from power sources on the station, on the nuclear shuttle, and perhaps from nuclear experiments; by requirements for astronaut visibility; by space station/base geometry; and by interference from the sun when in the field of view.

Most of these restrictions involve inter-program interfaces which will require timely exchange of information between programs, interface definition, and interface control.

3.3 DEPLOYMENT AND RETRIEVAL OF PAYLOAD

The revised study plan designates Task 2.2.6, Cargo/Passenger/Handling as a special emphasis task. In Section 4 a module payload concept is developed to satisfy all mission requirements with a minimum number of module configurations. The modules are then used in an analysis of deployment and retrieval techniques for each mission. The requirement for consideration of all missions in making a final selection of payload concepts and on-orbit handling techniques becomes apparent in this section.

3.4 ENTRY AND LANDING

3.4.1 ENTRY. Entry is effected upon completion of the normal orbital operations, or in case of an abort situation. The events during entry are:

a. Perform entry checklist.

b. Check weather at destination and alternate landing sites.

c. Pressurize interfaces with helium.

d. Maneuver vehicle into retro firing attitude with the attitude control system.

e. Settle propellants.

f. Start engines and modulate to required retro thrust pulse.

g. Verify retro impulse.

h. Yaw vehicle 180 degrees and assume required entry pitch angle.

i. Monitor entry attitude and temperatures.

j. Perform pitch change maneuvers as required.

l. Monitor lateral range.

m. Perform banking maneuvers as required.

n. Conduct glide to altitude at which terminal maneuver starts.

3.4.2 CREW ACTIVITIES DURING TERMINAL MANEUVER. After the velocity of the orbiter has slowed to subsonic, which will be at approximately 25,000 feet, the vehicle will be configured to conform to normal manned aircraft type of operation. The terminal maneuver and landing will be similar to any large cargo-type aircraft. The events during this phase follow:

3.4.2.1 Entry Recovery

a. Establish glide for engine start.

b. Extend engines and perform air start procedure.

c. Check controls and displays for system condition.

d. Contact FAA flight services via radio to report position per filed flight plan.

e. Establish cruise altitude per filed flight plan.

f. Establish relative heading to terminal.

3.4.2.2 En Route Activities

a. Check current geographic position —
   Use inertial navigation equipment and radio aids.
   Use radio DMI bearing to check and verify position.

b. Set course to destination and alternatives.

c. Report to FAA flight services in accordance with IFR or VFR flight plan.

d. Report progress and pertinent facts to home base — request support if alternative is necessary.

e. Perform onboard system checks, including fuel management.

3.4.2.3 Landing Duties

a. Contact terminal control for landing instructions and atmospheric conditions.

b. Establish approach configuration —
   High-lift devices in approach setting.
   Power set for desired rate of descent.

c. Report to airport control upon entering base leg to landing —
   Power and trim set for approach speed and rate of descent.

d. Report to airport tower upon turning to final approach (active runway heading) —
   Extend landing gear — check down and lock indication.
   Extend high lift/drag devices to landing position.
   Adjust power and trim for desired landing approach flight path.
   Proceed to touchdown.

e. Touchdown and roll out —
   Cut power on touchdown.
   Retract high lift devices.
   Use brakes, as weight is transferred from wing lift to landing gear, to decelerate to a stop or minimum taxi speed.

f. Contact airport ground control —
   Clear active runway and prepare to taxi or accept tow to terminal area.

SECTION 4
PAYLOAD DESIGN AND OPERATIONS

This section documents the results of space shuttle payload studies. The payloads and their deployment with the space shuttle are based on the mission requirements discussed in Section 2. During this study a modular payload approach was used to enhance space shuttle mission flexibility. The baseline space shuttle orbiter element is provided with a payload bay and mission-peculiar payloads are modularized for installation in this bay. Trade studies were made to compare this basic modular approach with integrating the payload into the basic space shuttle.

4.1 GENERAL PAYLOAD REQUIREMENTS

This section discussed the general approach to space shuttle payload design considering the missions described in Section 2. The remaining sections then describe payloads for individual missions in greater detail.

4.1.1 PAYLOAD IDENTIFICATION. Referring to the mission requirements of Section 2, the following types of payloads are considered:

a. Passengers and cargo

b. Propellant

c. Propulsive stages

d. Satellites

e. Maintenance module

f. Sensors and miscellaneous equipment

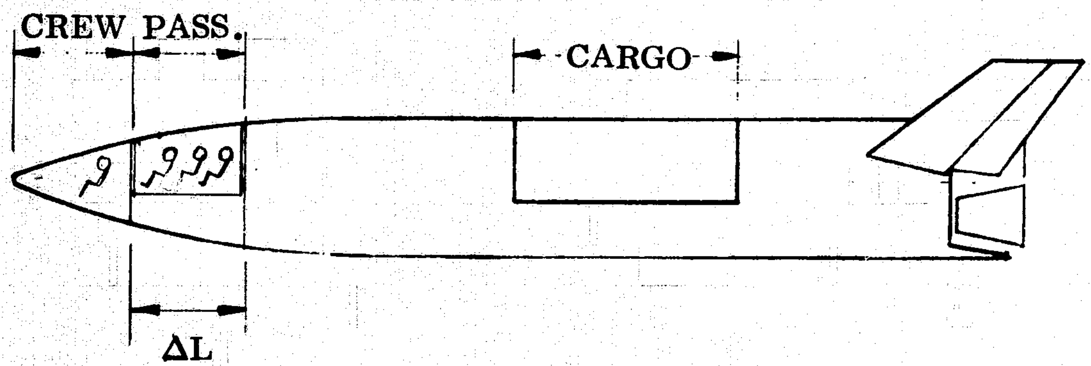
Layouts of typical payloads listed above indicated that a modular approach was feasible. The modular approach would offer the following advantages:

a. Minimize space shuttle ground turnaround time as mission peculiar modules could be quickly installed in the payload bay with a minimum of modification to the orbiter.

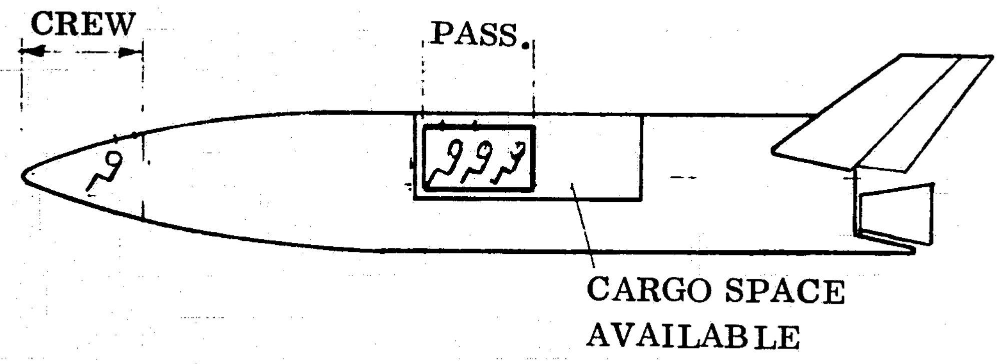
b. Maximize performance as mission-peculiar equipment would not be incorporated in the orbiter and carried on other missions.

c. Reduce the length of the orbiter. As illustrated in Figure 4-1, if the passenger section were moved forward and integrated into the basic orbiter airframe, the

INTEGRATED PASSENGER SECTION



MODULAR PASSENGER SECTION
(IN PAYLOAD BAY)



Human Factors	More conventional regarding closeness to captain	Physical confinement may create increased passenger anxiety
Safety	Advantage of single ingress/egress hatch	Possible disadvantage due to hatch in payload door as well as normal hatch on module
Mission Use Rate	Passenger section not filled to capacity on most missions	Passenger module removed when not required
Weight	Weight penalty on alternate missions (1600 lb ECS & LSS, 7000 lb structure)	No weight penalty on alternate missions (Approx. 500 lb less on passenger missions)
Length	Approximately 12 ft additional length	No additional vehicle length required
Complexity	Less complex for passenger missions (common life support subsystems)	Separate life support subsystems
Operations	Two docking ports required for unloading (or tunnel between passengers & cargo)	Common docking port for unloading passengers & cargo

Figure 4-1. Basic Passenger Arrangements

additional volume could increase the vehicle length up to 12.0 feet with four abreast seating. This is because the payload bay is sized by cargo and the passenger module fits in the available space.

d. The modular approach simplifies the interface control problems as clean interfaces can be defined between the basic orbiter and mission peculiar payloads.

The above considerations make the modular approach attractive for this phase of the program.

4.1.2 MODULE SIZING ANALYSIS. Payloads directly applicable to modularization are:

a. Passenger compartment

b. Cargo storage

c. Propellant storage

d. Service and maintenance facility

e. Experiment storage

The group of modules should have sizes compatible with the 60-ft payload bay and each other; i.e., module lengths of 15, 30, 45, and 60-ft long.

A brief analysis was conducted to first identify candidate personnel module sizes.

The mission requirements presented in Section 2 indicate the need for personnel in addition to the two-man orbiter crew to perform certain mission operations. In addition, other mission requirements include the delivery of personnel to in-orbit vehicles or manned satellites. All missions require that these additional personnel be provided with a shirtsleeve environment. Figure 4-2 summarizes the size requirements for the personnel modules based on the Section 2 mission model.

In an effort to reduce total costs associated with these modules, a limited number should be developed to handle all mission requirements. The five sizes are functionally separable. The 6, 10, and 12-man sizes are primarily delivery missions, with the 2 and 4-man sizes associated with special personnel for on-orbit operations. The two module sizes which appear most desirable are therefore a 12-man module and a 4-man module. The 12-man module, offloaded for 10- and 6-man missions, would be used on 210 missions. The 4-man module, offloaded for 2-man missions, would be used on 325 missions.

The 4-man module is discussed further in the following section. The 12-man module is defined in detail in Section 4.2.1.

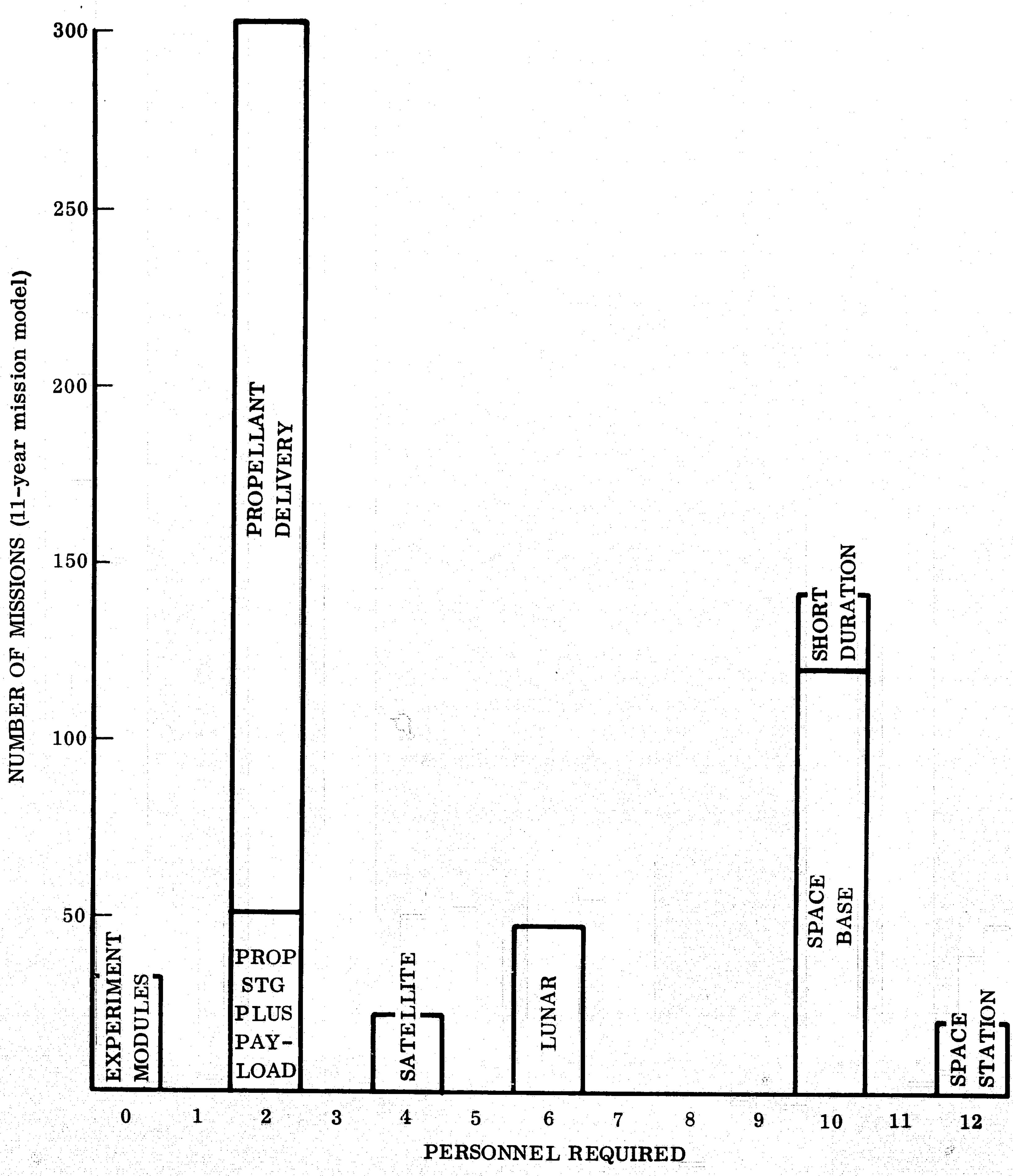


Figure 4-2. Personnel Module Size Requirements

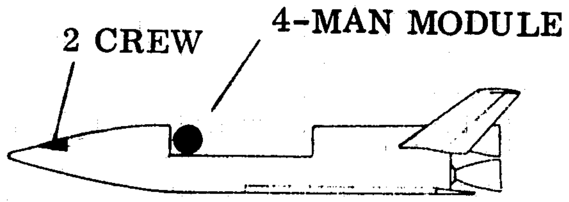
4.1.3 COMPARISON OF 4-MAN MODULE AND 6-MAN ORBITER. The 4-man personnel module is used on 325 (57 percent) of the missions. This module introduces both a payload compartment volume reduction and weight reduction due to the structure and subsystems required. As shown in Figure 4-3, an alternative to the 4-man personnel module would be a 6-man orbiter cab. This provides the necessary personnel requirements, without payload volume being reduced by 17 percent, and a probable reduction in weight associated with supporting subsystems. An increase in size of orbiter subsystems such as EPS, LSS, and ECS would take the place of the duplication of these systems when a separate module is used.

Use of the 4-man module has some advantages such as direct visibility for orbital operations. The 4-man module is used for personnel responsible for placing and retrieving satellites, delivering propellant, and delivering propulsion stages and payloads. If these operations are performed from the orbiter cab, indirect viewing such as TV would have to be provided or an aft control station would be required. The module or aft control station would require a computer for checkout functions or use the orbiter avionics on a time-share basis.

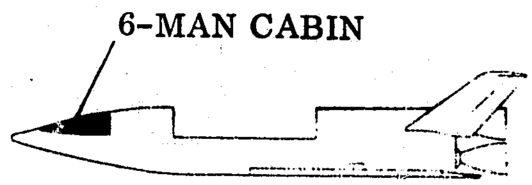
If a 6-man personnel (including crew) capability were designed into the orbiter, it could also be used for these missions requiring a 12-man module. The result is that the 12-man personnel module could be sized for eight men at 10 feet in length. It is estimated that this module would weigh 7200 lb. When used as a cargo module, it would weigh about 5000 lb. The orbiter 6-man cab capability would now be used for 94 percent of the missions. The only exception is the experiment module delivery missions, which will be discussed below.

A set of typical experiment modules is presented in the NASA Space Shuttle Task Group Report, Vol. 1, 12 June 1969. Figure 4-4 relates the diameter and weight of these modules to the length. All modules are 15 feet or less in diameter and weigh 32,000 pounds or less. Of the 17 modules shown on the length/diameter cross-plot, six are less than 30 feet in length. These six modules could be part of a normal logistics mission when using a 12-man personnel module and cargo module. The use of the 8-man personnel module allows 13 of the 17 modules to be carried during normal logistics missions. The result is the reduction of the number of separate experiment module flights. The mission model shows 33 flights. If module sizes retain their current distribution, only 24 percent of the modules would require separate flights. This means that the orbiter cab would be used for 98.5 percent of the flights.

4.1.4 COMBINED 8-MAN PERSONNEL/CARGO MODULE. The 8-man personnel module is used for 210+ flights. (The + represents rescue missions.) The cargo module is used for 188 missions. The difference is the 22 short-duration orbital missions. A combination of the two modules will save approximately 2.2 feet of length and some weight by removing two bulkheads and hatches. Additional weight is saved by reduction to one EPS, one LSS, one H_2O, one ECS, one communication



Direct viewing of payload
Used on 46% of flights
Payload length reduced 10.0 feet
Minimizes orbiter length
Liftoff weight reduced on alternate missions



Indirect viewing by TV or direct by tunnel
Used on 98.5% of flights
Possibly adds to orbiter length
Liftoff weight reduced on missions requiring 4 men*

*Approximately 400 lb saving in integrated environmental control and life support subsystems.

Figure 4-3. Alternate Location — 4 Men

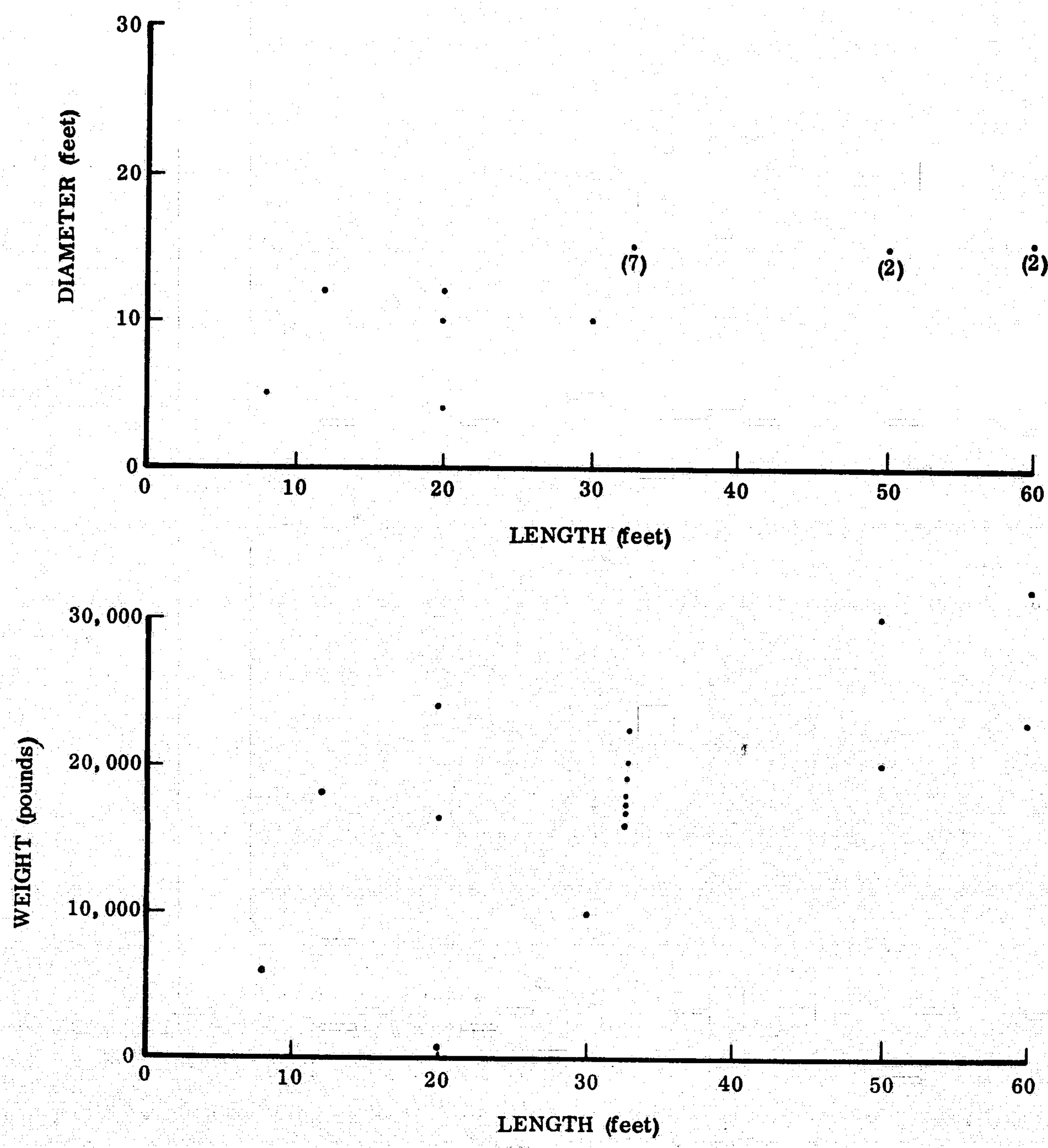


Figure 4-4. Experiment Module Relationships

subsystem, and one cryogenic subsystem. These single subsystems will also result in usable internal volume increase. In addition, a five-foot-diameter and a three-foot-diameter hatch can be eliminated, with subsequent weight saving and usable volume increase.

4.1.5 SUMMARY OF MODULE REQUIREMENTS. The payload identification and sizing considerations discussed in the previous sections can be summarized as follows:

a. When "mission peculiar" personnel and/or equipment are modularized, a series of modules will be required similar to those summarized in Figure 4-5. Preliminary module design requirements are summarized in Table 4-1.

b. When the orbiter stage cab is increased in size to accommodate four mission personnel, the mission modules would be similar to those summarized in Figure 4-6. The shaded area represents increased payload length compared with using a 4-man module.

   A new set of module requirements reflecting this configuration is shown in Table 4-2. The results are increased cargo length of 10 feet in 57 percent of the flights and five feet in 37 percent of the remaining missions. Additionally, weight savings should be apparent due to the elimination of the 4-man module on many flights.

c. Combining the 8-man personnel module and the cargo module as discussed in Section 4.1.4 appears attractive as it results in a length reduction of 2.2 ft and reduction of module weight and complexity. Table 4-3 summarizes the module arrangements when the combined personnel/cargo module is used.

On the basis of the studies to date, option b above appears most attractive. The basic reason is the increase of mission flexibility of the space shuttle; i.e., more experiment modules can be delivered when mission support personnel can be carried in the orbiter cab. Based on information to date, the additional number of passengers would be up to four and assuming two orbiter crew members, the maximum requirement is six. Additional crew function analyses of each mission may allow the total number to be reduced to four or five.

The question of combining modules as suggested in c above requires additional system analyses. Converting modules from passenger to cargo is discussed further in the next section.

4.2 SPACE STATION/BASE LOGISTICS

This section documents the baseline space station/base payload module developed during the study. Several experiment modules under development are also shown because they are candidate payloads. The docking and orbital operations are also discussed to derive space shuttle requirements.

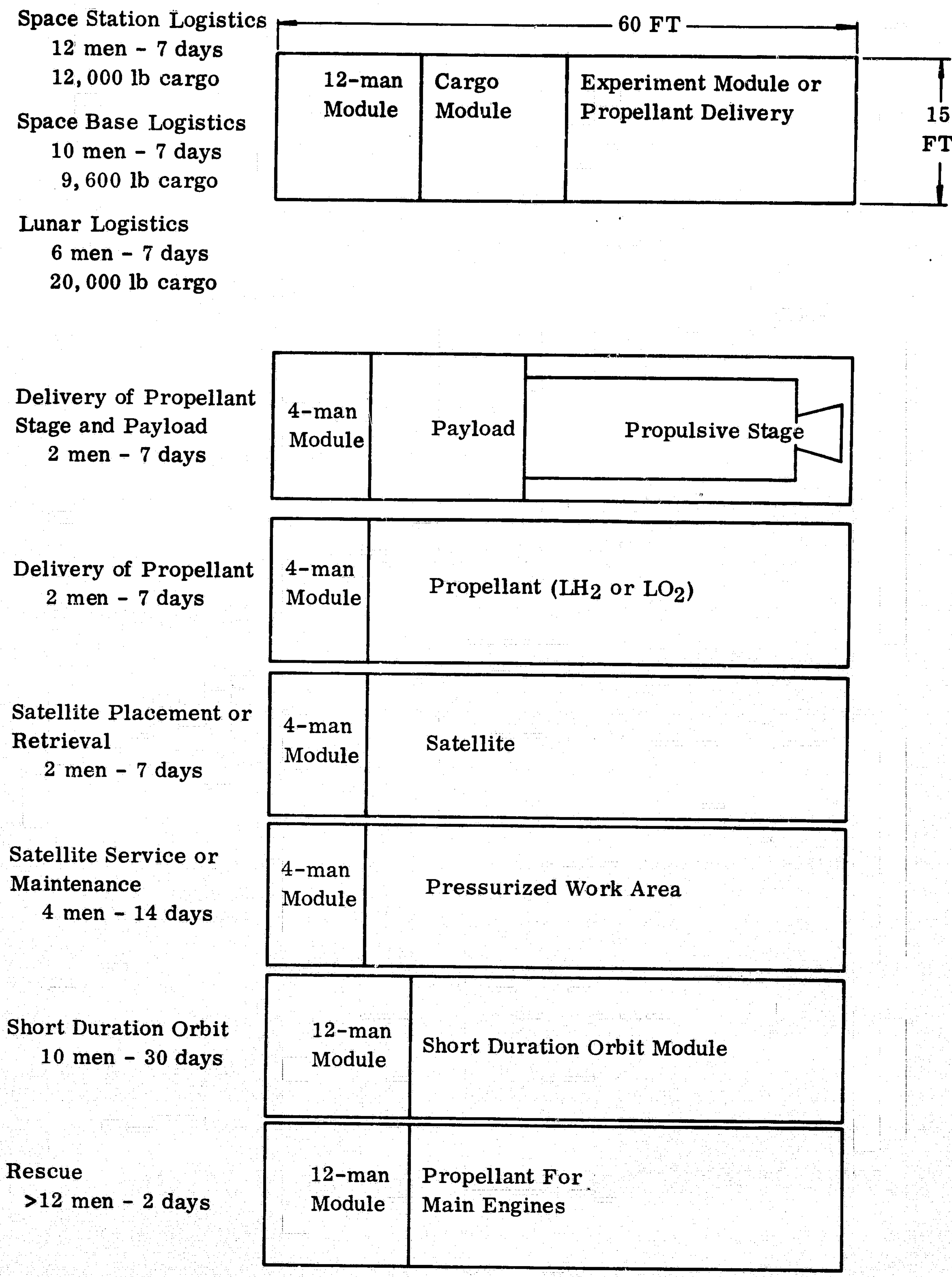


Figure 4-5. Mission Module Applications

Table 4-1. Mission Module Requirements

Module	Pressurized	Life Support		Length (feet)	Number of Times Used	Comments
		Men	Days			
4-man Personnel	Yes	4	7	10	325	Direct view of payload bay
12-man Personnel	Yes	12	7	15	210+	See Section 4.2 for detailed design
Cargo Module Converted from 12-man Module	Yes	-	-	15	188	9,600 lb space base 12,000 lb space station 20,000 lb lunar
Small Propellant	No	-	-	30	<210	LH_2 or LO_2
Large Propellant	No	-	-	50	252	LH_2 or LO_2
Pressurized Work Area	Yes	6	7	50	<22	
Short-Duration Orbit	Yes	10 2	30 23	45	22	Possible use as pressurized work area
Experiment	Module Dependent	Module Dependent	Module Dependent	<30	33	60-ft length when not transported with personnel and cargo module

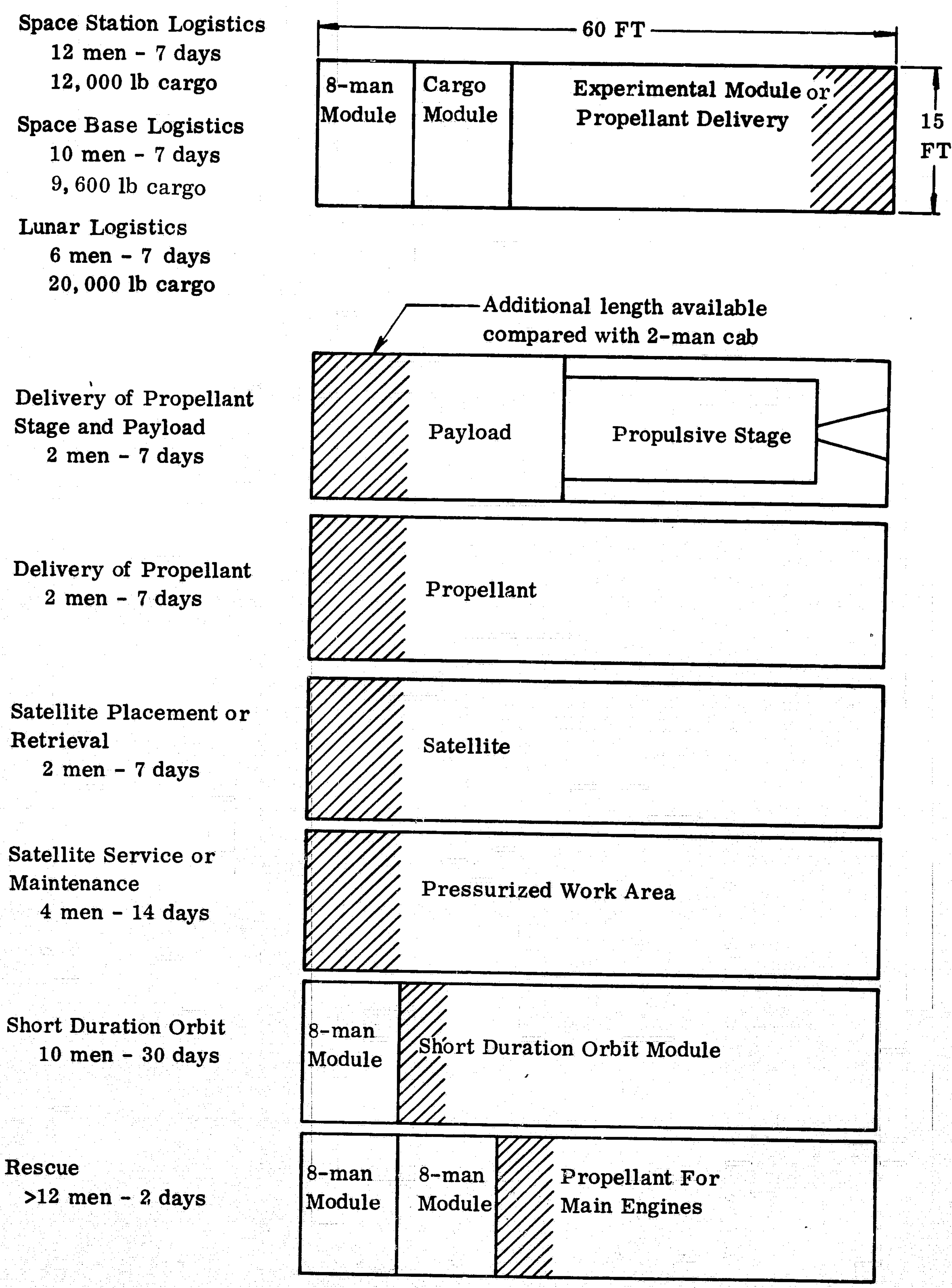


Figure 4-6. Mission Module Applications With 6-Man Orbiter Cab

Table 4-2. Module Requirements Using 6-Man Orbiter Cab

Module	Pressurized	Life Support		Length (feet)	Number of Times Used	Comments
		Men	Days			
8-man Personnel	Yes	8	7	10	210	
Cargo Module Converted from 8-man Module	Yes	-	-	10	188	
Small Propellant	No	-	-	40	<210	LH_2 or LO_2
Large Propellant	No	-	-	60	252	LH_2 or LO_2
Pressurized Work Area	Yes	6	7	60	<22	
Short-Duration Orbit	Yes	12	23	50	22	Possible use of pressurized work area
Experiment	Module Dependent	Module Dependent	Module Dependent	<40	<33	60-foot length not transported with personnel and cargo module

Table 4-3. Module Requirements Using 6-Man Orbiter Cab and 8-Man Personnel/Cargo Module

Module	Pressurized	Life Support		Length (feet)	Number of Times Used	Comments
		Men	Days			
8-man Personnel/ Cargo	Yes	8	7	17.8	210	Can be used for over 18 people or all cargo
Small Propellant	No	-	-	42.2	<210	LO_2 or LH_2
Large Propellant	No	-	-	60	252	LO_2 or LH_2
Pressurized Work Area	Yes	6	7	60	<22	
Short-Duration Orbit	Yes	12	23	42.2	22	
Experiment	Module Dependent	Module Dependent	Module Dependent	<42.2	<33	60-foot length not transported with personnel and cargo module

4.2.1 BASELINE MODULE. A convertible 12-man personnel/cargo module was defined to support the payload study. The convertible 12-man personnel/cargo module requires a flexibility to handle the logistics for a 12-man space station of 12,000 pounds and 12 men (up) and 7,000 pounds and 12 men (down) quarterly, a 50-man space base of 9,600 pounds and a 10 men (up) and 5600 pounds and 10 men (down) every 18 days, or a lunar mission of 6 men and 20,000 pounds every two months. The types of cargo include food, experiment modules (docked and free flying), liquids, operational equipment, gases, personnel, and spares. Cargo mixture may include cargo, personnel, experiment modules, and propellant deliveries on the same flight.

A preliminary layout was performed to verify the feasibility of carrying 12 men in a container 15 feet in diameter by approximately 15 feet long (Figures 4-7 and 4-8). Ground rules used to generate the concept are:

a. The shuttle would be hard-docked to the space station or space base.

b. Transfer connection to the station or base would be via an extending tunnel originating on the station or base. The hatch does not have docking provisions. This permits the lightest weight module and hence greatest useful payload.

c. All hatches will be five feet in diameter, clear.

d. All parts of system will have maximum reusability (convertible to cargo).

e. Airline-type system operation will be used (convertible to cargo).

f. The module will not be removed from the shuttle bay.

g. Cargo can be mixed internally.

h. Cargo and passenger loading and unloading can be performed directly.

i. Intact mission abort capability will be designed in.

j. Personnel transfer to base or station will be "shirtsleeve".

The probable operational mode is to carry 6 to 12 passengers on a mission of one day up, one day back, and four to five days at the station or base for loading, reloading, or standby. Salient features of the module are:

a. Self-sustaining (only interface to shuttle vehicle is at physical attachment to longeron).

b. Meteoroid protection open end only. Shuttle vehicle provides remainder of protection inherently.

c. Seats only, no bunks.

d. Five-foot-diameter tunnel with hatch for connection to docked vehicle such as space station or space base.

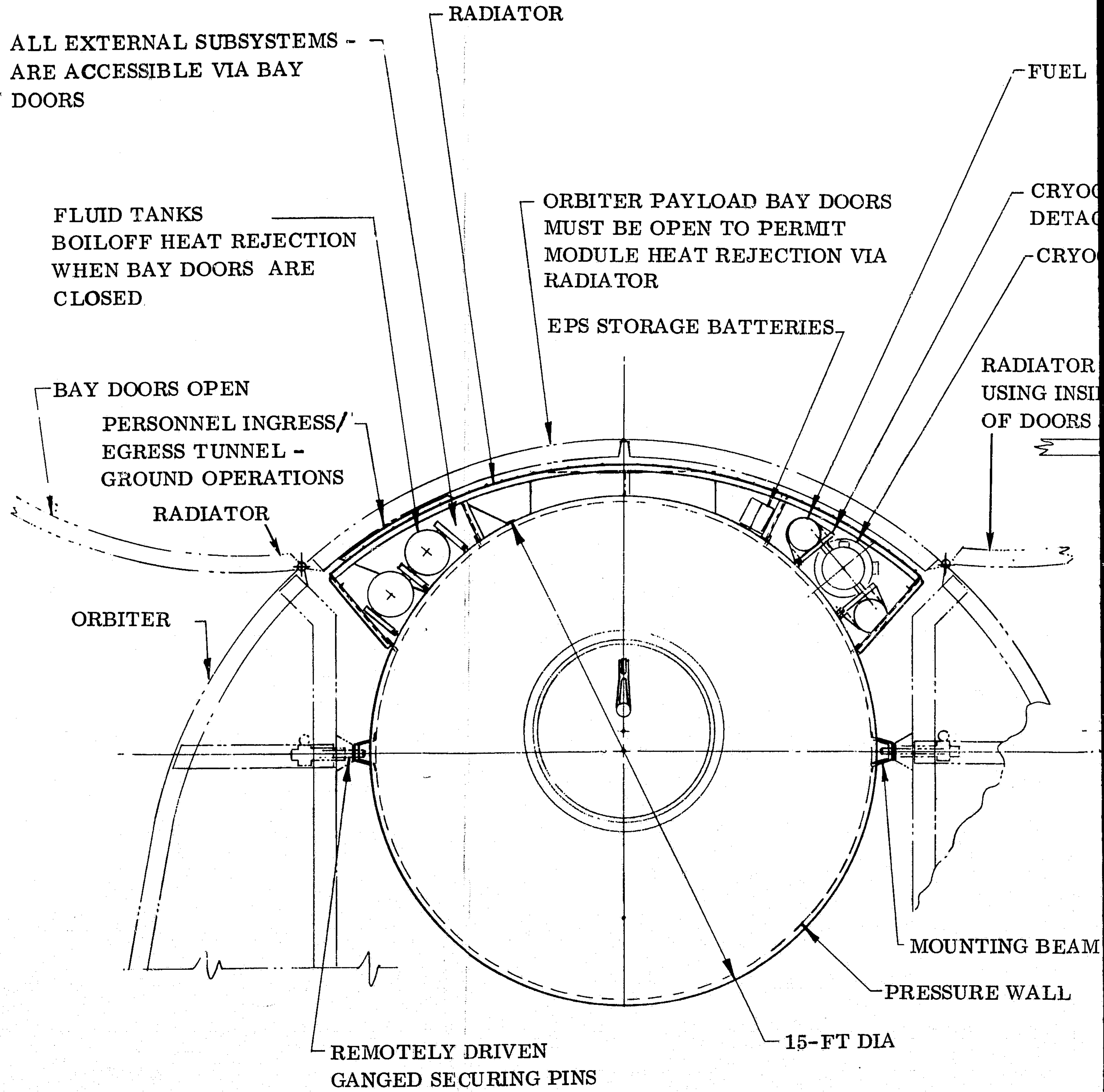
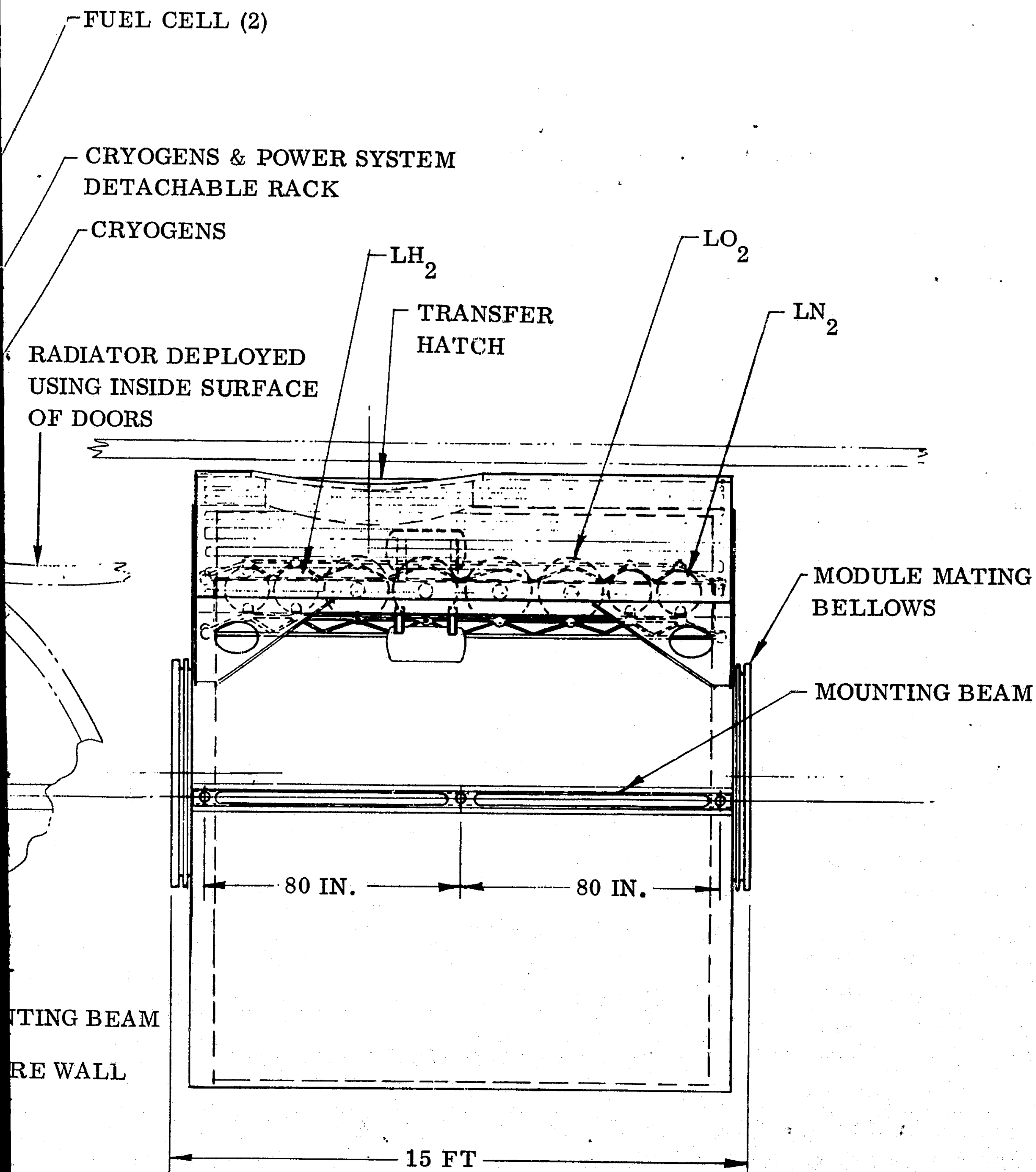
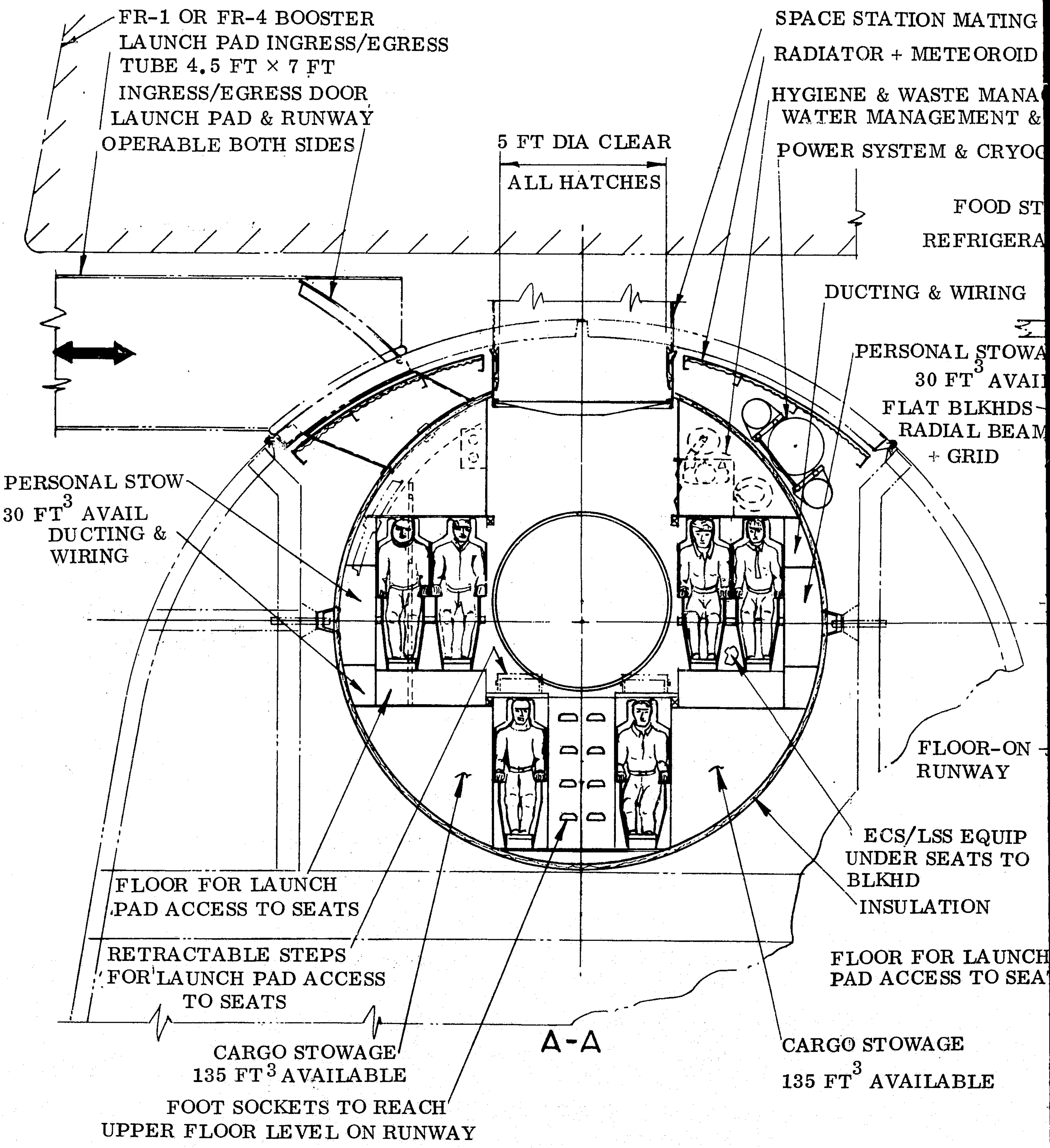


Figure 4-7. 12-Passenger Self-Sustaining Module — External Configuration



FOLDOUT FRAME 2



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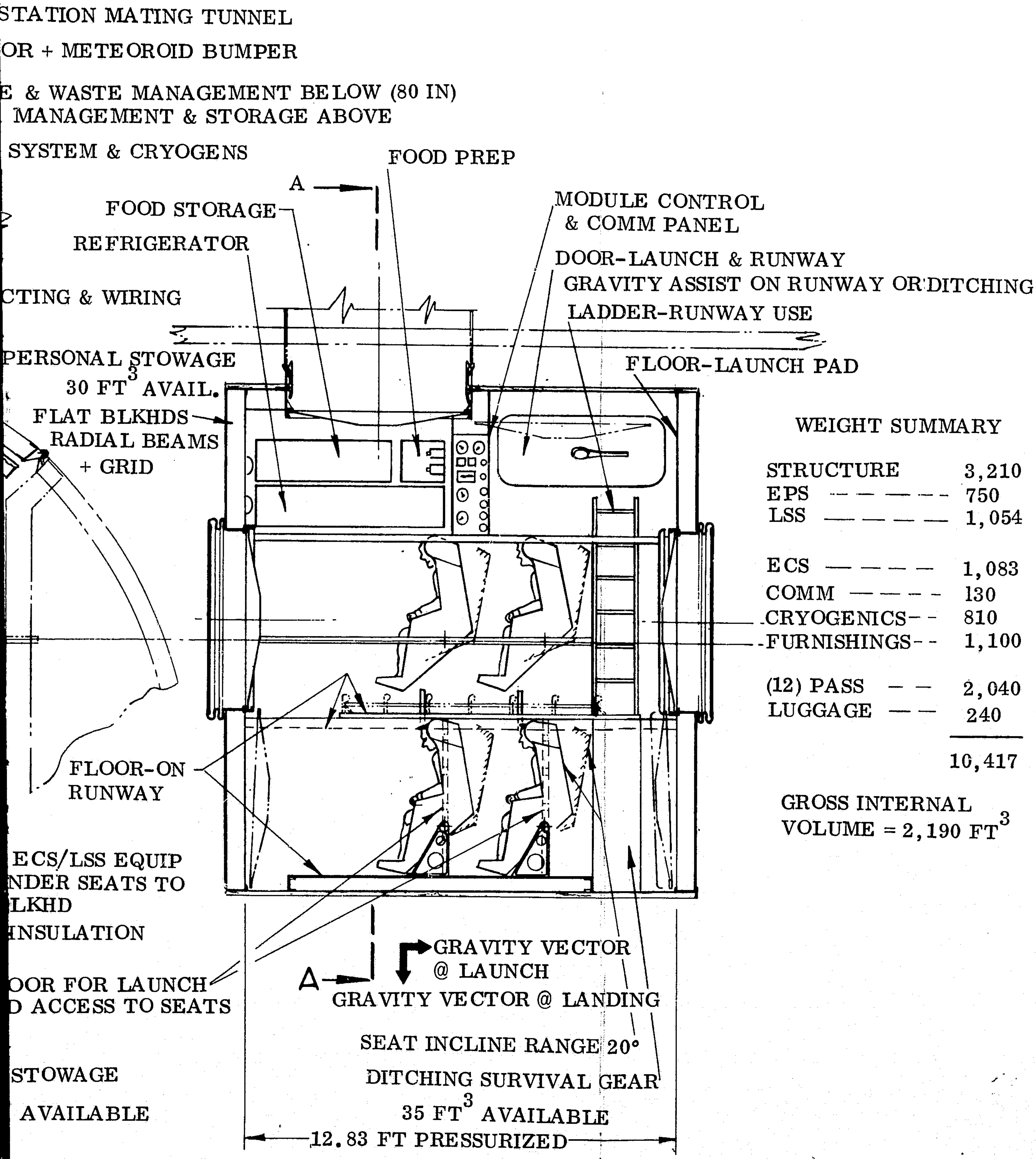


Figure 4-8. 12-Passenger Self-Sustaining Module — Inboard Profile

e. Five-foot-diameter tunnel with hatch for interconnection to similar personnel or cargo modules.

f. Center core unobstructed to permit free passage of personnel or cargo to limit of hatch sizes.

g. Removable seats to convert to cargo hauling in controlled environment or mixed material/personnel cargo.

h. Change from passengers to cargo or cargo to passengers can be accomplished in orbit as well as on ground.

i. No airlock.

j. Water ditching provisions.

k. Quick evacuaton on pad.

l. Mixed gas atmosphere at 10 psia.

m. LSS/ECS sized for 12 men for seven days.

Consumption rates:

Food consumption	2 lb/man/day
Metabolic O_2 used	1.68 lb/man/day
H_2O consumption, drinking and food prep.	6.99 lb/man/day
Personal sanitation H_2O	2.3 lb/man/day
Atmosphere leakage allowance	10 percent gross vol/day

No O_2 recovery

No H_2O recovery from waste

CO_2 removal by LiOH

Odor removal by activated charcoal

Heat rejection radiator, H_2O/ethylene glycol

Heat rejection on pad and entry by fluid boiloff

n. EPS

Two fuel cells used for primary power and redundancy. Each rated at 1.5 kW continuous, 2500-hour life.

Estimate continuous power level is 1.0 kW

o. Cryogenic Stores

Supercritical storage

Usage Temperature = 40°F

Combined LSS and EPS storage

Integrated requirements for 12 men, seven days are shown in Table 4-4. Table 4-5 presents a weight summary by subsystem.

Table 4-4. Cryogenic Stores Summary

Activity	Unit Usage	Factor	O_2	N_2	H_2
Module Pressure	33.3 lb O_2/pressure	1 time	33.3		
	75.7 lb N_2/pressure	1 time		75.7	
EVA	None	-	-	-	
Module Leakage	3.3 lb O_2/day	7 days	23.1		
	7.6 lb N_2/day	7 days		53.2	
Metabolic O_2	20.2 lb O_2/day	7 days	141.4		
Power	24.4 lb O_2/day	7 days	169.4		
	3.0 lb H_2/day	7 days			21.0
Total Weight (lb)			367.2	128.9	21.0

Typical module design problems are:

a. Hatches. The five-foot-diameter hatch presents a problem of removal and storage. Each hatch, which weighs approximately 120 pounds, must be restrained and guided to stowed position. Stowed position requires considerable volume. Hinged or swinging hatches sweep out excessive internal volume. It was necessary to offset bulkhead hatches from module centerline to permit stowage without taking excess volume.

b. Environmental Control. Preliminary estimates to remove excess heat generated by 12 people and electrical equipment when enclosed behind exterior doors show that 1800 pounds of water would be required to boiloff the total heat load. Design of radiators in shuttle skin is complicated due to materials and the high skin temperature (over 700°F) at entry.

c. Floor Arrangement. The gravity vectors at takeoff and landing are 90 degrees to each other. Floors at launch are walls when landing; hence, walking surfaces and ladders are necessary. This factor influences abort capabilities relative to ease and speed. The single ground access door used becomes inclined 90 degrees after landing. In the FR-1 configuration, a special ground-use door is necessary because of the stacked configuration on the pad.

Table 4-5. Weight Summary by Subsystems

Subsystem		Weight, lb
Structure		3,210
Primary	2244	
Secondary	507	
Hatches and Doors	449	
EPS		750
Fuel Cells (2)	136	
Batteries	50	
Conditioning and Distribution	564	
LSS		1,054
Atmosphere Control	581	
Food and Processing	173	
Water Management	163	
Waste Management	105	
Personal Hygiene	32	
ECS		1,083
Cabin Control	335	
Radiator System	658	
Insulation	90	
Communications		130
Cryogenic Stores		810
Expendable Gases	517	
Tankage	192	
Plumbing, Controls, Supports	101	
Furnishings		1,100
Seats (12)	1040	
Attachments, Padding, Holds	60	
		8,137
Payload (Useful)*		2.280
Passengers (12)	2040	
Luggage	240	
Total Module Weight		10,417

*135 ft^3 of cargo storage available

d. Cargo Handling. The problem is to achieve an arrangement which permits ease of access and to avoid clogging by bulky items. Also, emergency priorities may develop between ground loading and orbit transfer. When loading on the ground or in orbit, the back sides of packages against walls are generally inaccessible. Therefore, securing of cargo must be accomplished from the front face. This is achieved on the example module by keeping the central core clear.

A variation of this module has no hatch in the forward (airplane reference) bulkhead. Instead, a 32-inch-diameter hatch is installed in the lower forward cylindrical surface and mated to the tunnel connecting the module to the cockpit. This module would be used much more frequently than that shown in the figures since it is always needed for interconnection to the cockpit. No preliminary layouts were made because of its similarity to the baseline design.

The pressurized cargo module does not require design or manufacture of a specific module. In the payload concept exampled, the personnel module is convertible to a full cargo configuration (Figure 4-9). The basic subsystems of the personnel module are retained to provide power, environmental control, and atmosphere control. The seats are removed as well as the food storage and preparation buffet. Cryogenics and H_2O can be offloaded to match the cargo requirements. A weight breakdown of subsystems is shown on Figure 4-9, reflecting a total cargo module. Overload capability or outsize cargo can be accommodated volumetrically.

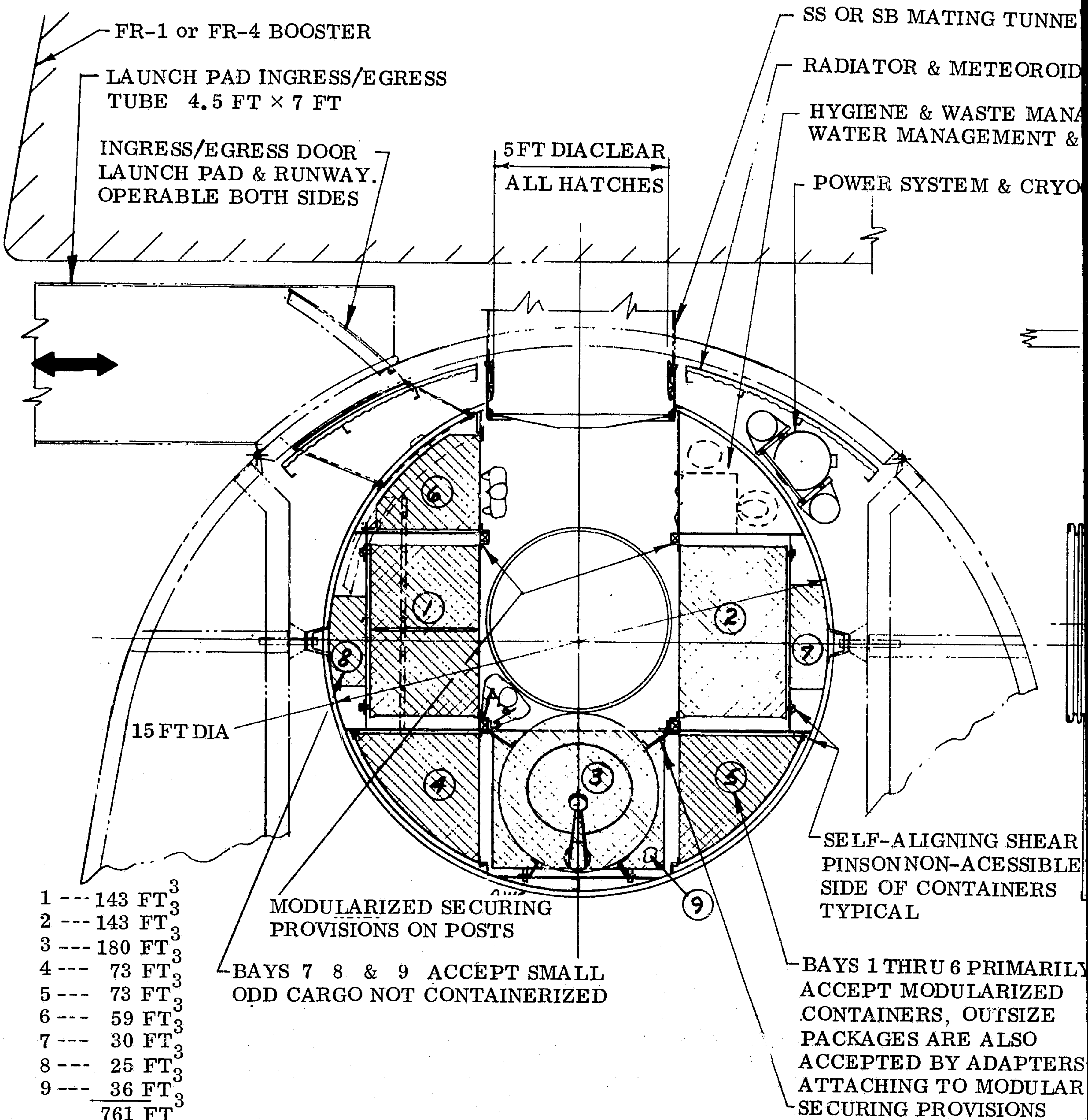
An analysis was conducted to verify the desirability of only a mechanical interface. Table 4-6 shows the 12-man module subsystem weight breakdown. The first column reflects the recommended mechanical interface and shows a total module weight, including passengers and luggage, of 10,414 lb.

If an electrical interface is added, the fuel cells and fuel will be moved to the orbiter. The orbiter power output will then be increased to 5.7 kW from 4.5, and there will be an increase in weight in the conditioning and distribution system. In addition, the backup battery can be removed. The weight savings for the module will be approximately 1000 lb with a weight increase of approximately 500 lb on the orbiter. This is reflected in columns 2 and 3. Column 4 indicates an approximate 500 lb weight savings for the logistics mission with one module for an electrical interface. The 500-lb orbiter weight increase is a weight penalty for those alternate missions not requiring an electrical interface. The penalty to the orbiter electrical system increases if it must be sized to handle four 12-man personnel modules, which represents a doubling of the original power requirements. Even if this is done there is no guarantee that adequate electrical power is available for alternate missions.

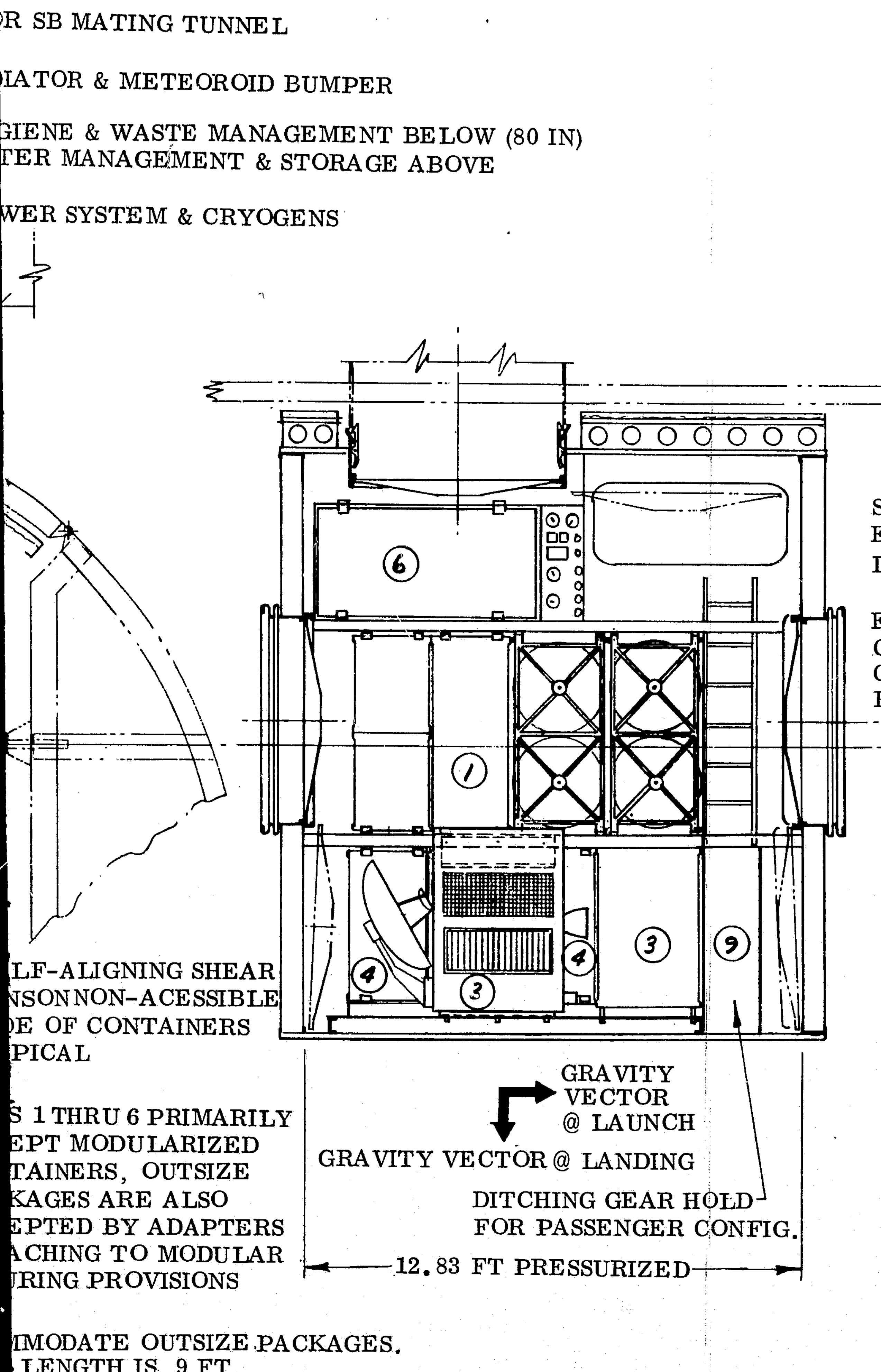
Similar effects are noted in columns 5, 6, and 7 for a mechanical, electrical, environmental control system (ECS), and life support system (LSS) interface. Although the single module weight decreases, the orbiter is penalized. Sizing is again a problem

Table 4-6. Twelve-Man Personnel Module Interface Analysis
(Based on Orbiter Carrying One Personnel Module)

Subsystems	Interface						
	Mechanical	Mechanical and Electrical			Mechanical, Electrical, LSS, and ECS		
	Module Weight (lb)	Module Weight (lb)	ΔOrbiter Weight (lb)	Total Weight Savings (lb)	Module Weight (lb)	ΔOrbiter Weight (lb)	Total Weight Savings (lb)
Structure	3,210	3,210			3,210		
EPS Fuel Cells	136	--	+136		--	+136	
Batteries	50	--		50	--		50
Conditioning and Distribution	564	--	+94	470	--	+94	470
LSS Atmosphere Control	581	581				<+581	
Food and Processing	173	173			173		
Water Management	163	<163		--		<+163	
Waste Management	105	105			105		
Personal Hygiene	32	32			32		
ECS Cabin Control	335	<335		--		<+335	
Radiator System	658	658			658		
Insulation	90	90			90		
Communications	130	130			130		
Cryogenic Stores - Expendable Gases	517	327	+190		327		
Tankage	192	120	+72		120		
Plumbing, Controls, Supports	101	101			101		
Furnishings	1,100	1,100			1,100		
Payload - Passengers	2,040	2,040			2,040		
Luggage	240	240			240		
Total Weights	10,414	<9,402	+492	520	8,323	<1,310	520+



NOTES: 1. 470 FT^3 IS AVAILABLE THROUGH CENTER SECTION TO ACCOMMODATE OUTSIZE PA
2. LONGEST POSSIBLE LENGTH = 11 FT. LONGEST CONVENIENT LENGTH IS 9 FT.



WEIGHT SUMMARY

STRUCTURE	2,960
EPS	600
LSS	636
ECS	1,083
COMM	130
CRYOGENICS	360
FURNISHINGS	80
	5,849
PAYLOAD	12,000
	17,849

GROSS INTERNAL

VOLUME = 2,190 FT^3

Figure 4-9. Cargo Module Converted From 12-Man Module — Self-Sustaining, Pressurized

FOLDOUT FRAME 2

if capacity for four 12-man modules is required as the LSS/ECS weight is roughly proportional to capacity. The orbiter would then have a system which is approximately five times its nominal capacity. In addition the interface now has to consider transfer of fluids and gases between the orbiter and each of the modules in parallel or from the orbiter to the forward module and then between modules in series. For passenger and cargo transfer concepts utilizing module removal this more complex interface is not desirable.

The use of mechanical interface is recommended. It allows the orbiter to be sized to firm requirements and payloads to be sized to their unique mission requirements as they are developed. A mechanical interface is also the most desirable when considering personnel and cargo transfer concepts.

4.2.2 EXPERIMENT MODULE. A study to develop experiment modules is currently underway at Convair. Figure 4-10 shows a typical module being studied. Two important features are that the free flying module has both an attitude control system and a docking mechanism. The propulsion submodule is required to provide the necessary acceleration for the experiments. Docking with the space station is required for experiment servicing and propellant resupply.

4.2.3 DOCKING AND ORBITAL OPERATIONS. The logistics mission requires docking the orbiter with the space station and/or experiment modules. This section discusses orbiter docking concepts and orbital operations associated with docking.

4.2.3.1 Docking Requirements. The general requirements for docking systems are to:

a. Reduce the relative angular and lateral velocities to zero.

b. Minimize the impact loads.

c. Correct angular and lateral misalignments.

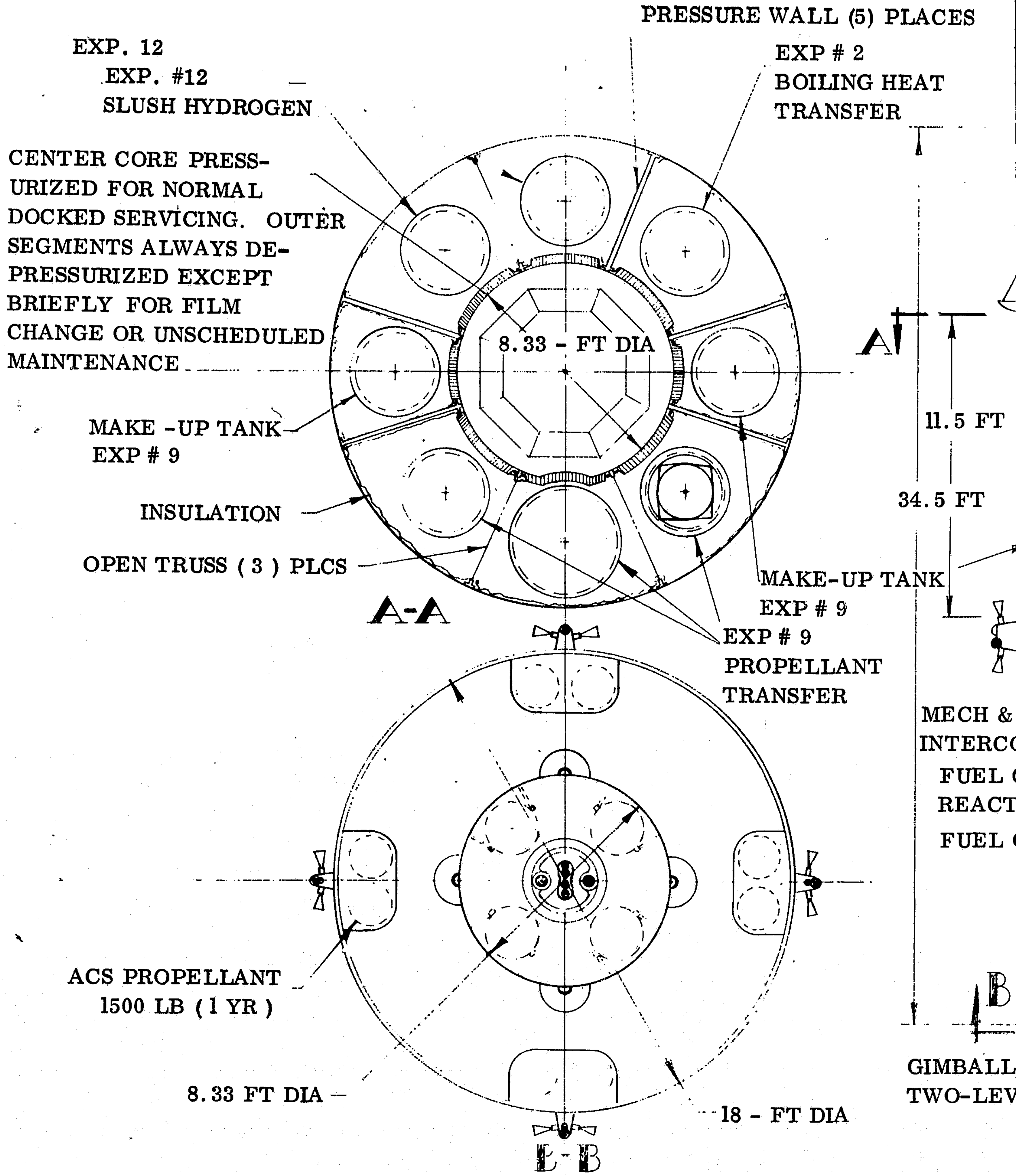
d. Secure the vehicle to the space station.

e. Seal the area around the transfer hatch.

f. Release the vehicle for the return trip

The parameters selected for docking based on space station studies are:

Axial Relative Velocity	2 fps
Lateral Relative Velocity	1 fps
Angular Relative Velocity	1 deg/sec
Lateral Miss Distance	12 in.
Angular Misalignment	10 deg



FOLDOUT FRAME 1

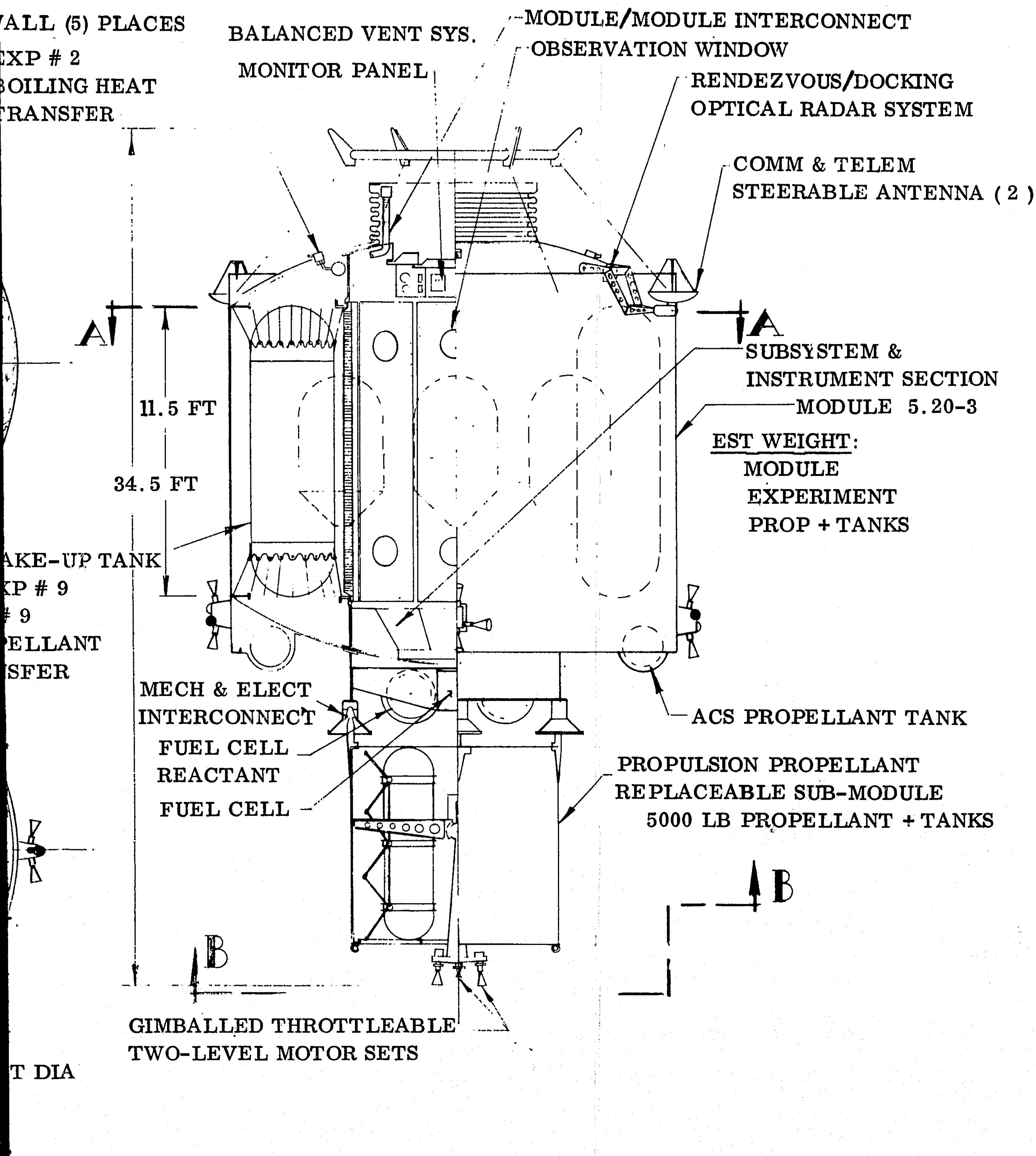


Figure 4-10. Fluid Physics Module

Functional requirements of docking systems have resulted from consideration of the advanced mission requirements experience gained during the Gemini and Apollo Programs. The advantage of an androgynous system, in which each spacecraft has identical docking hardware, is to allow a maximum degree of flexibility in mission definition and planning. Androgynous, in this case, means that the docking system of each vehicle can perform any mechanical function required for docking or separation without active aid from the docking system of the other vehicle. The docking-hardware configuration would no longer be a constraint on mission operations. The androgynous system also provides inherent redundancy, which is unobtainable from normal "male-female" type systems.

Good dynamic performance is not simply a desirable characteristic. It will be a definite requirement if one docking system is to be able to accommodate the wide range of vehicle configurations and weights involved in a space station program. Off-center docking will require a mechanism with a high degree of compliance to avoid structural dynamics problems and to allow capture latching to be readily obtained during impact. The "compliance" of a mechanism is the ability of the mechanism to compensate for misalignment between the docking vehicles at initial contact and to avoid rebound. High compliance is achieved by providing long-stroke energy absorbers and by avoiding the storage of large amounts of energy in undamped springs.

Because of the requirement for routine transfer of crewmen and/or cargo from one vehicle to another through a pressurized interface, it will be mandatory to avoid removal of the docking mechanism from the interconnecting tunnel. Therefore, the docking mechanism for advanced missions will be located outside of the pressurized-interface area. The interface pressure seal and the structural interconnecting mechanism will be separated so that the function of one will not affect the function of the other. The two basic reasons for this separation are as follows. First, separation will avoid the problem of the distortion of the sealing interface as a result of high structural preloading of the latch and as a result of differences in preloading from one latch to another. Second, the proper location and arrangement (diameter, etc.) of each system can be optimized independently with a minimum interaction between functions.

4.2.3.2 Docking System Location. Figure 4-11 shows three alternate docking arrangements. The concepts are based on the orbiter hard docking to the space station. Concept A of Figure 4-11 shows the location of the docking mechanism in the nose section of the space shuttle. The advantages of this concept are:

a. The pilot can perform the docking maneuver without leaving his normal flight position.

b. A separate docking station with console and crew member is not required.

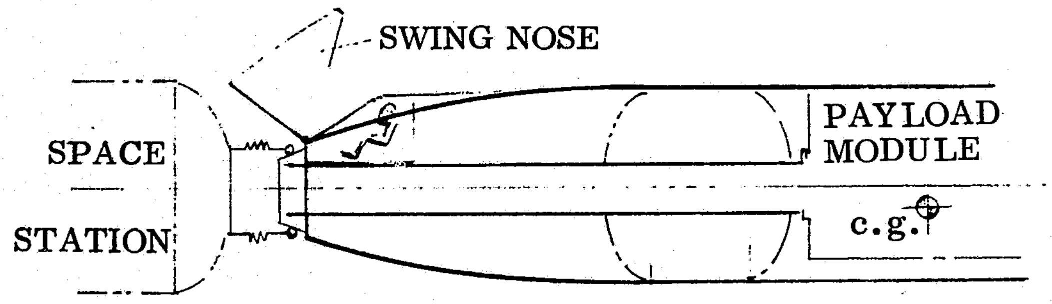
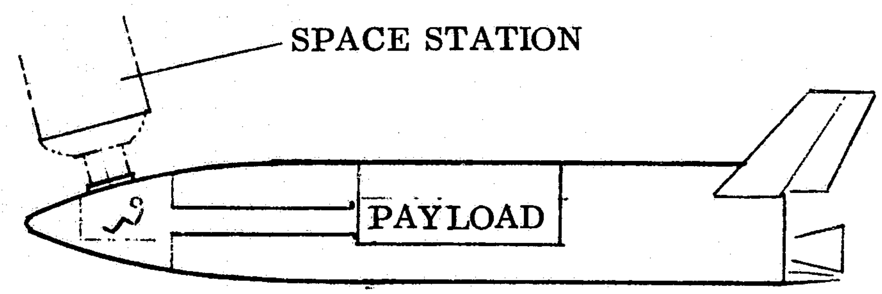
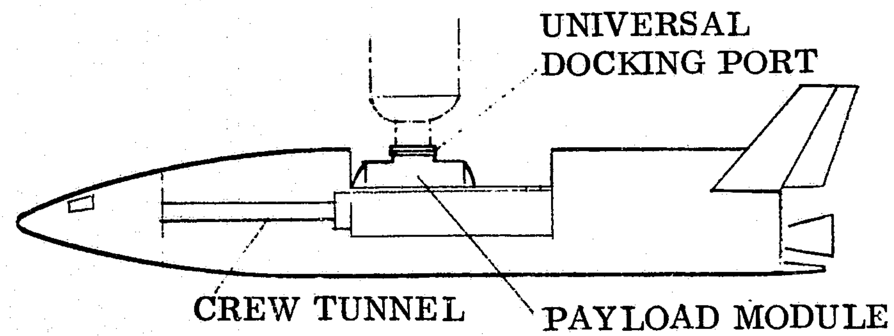
	ADVANTAGES	DISADVANTAGES
CONCEPT A - NOSE DOCKING  	Docking can be accomplished by pilot (separate control console not required)	Access tunnel required Reduced volume efficiency Docking port part of basic vehicle (mission peculiar item) Removable nose (may affect nose shape) Slow cargo unloading
CONCEPT B - CREW STATION DOCKING  	Same as above except: Removable nose not required	Pilot's view of docking not as good as Concept A's
CONCEPT C - PAYLOAD MODULE DOCKING  	Docking port made part of payload (mission peculiar) Low temperature location Most efficient concept	No direct view of docking by pilot; may require TV display or docking control at payload area

Figure 4-11. Vehicle Docking System Location

The disadvantages are:

a. A long, relatively large cargo transfer tunnel between the payload area and the nose section requires added volume and weight.

b. This location requires a removable nose cone and may affect the nose shape.

c. The docking port is part of the basic vehicle and will be carried on all missions.

Concept B shows the docking port relocated to the area of the flight deck. The advantage of this concept is that a swing nose is not required.

The disadvantages are similar to A; in addition:

a. The pilot's orientation and visual reference during docking is impaired and a special docking control station and/or display may be required.

b. The docking port is in a poor location regarding the flow of cargo and passengers from the payload bay to the docking port.

Concept C locates the docking port in the payload bay area. The advantages are:

a. The docking port can be made part of the payload modules and will not be carried on missions not requiring docking.

b. The existing payload bay doors will cover the docking mechanism and separate doors will not be required.

c. The location is ideal for unloading cargo and a cargo transfer tunnel is not required. (A smaller tunnel is required to provide the crew access to the payload area.)

The disadvantage is:

A separate docking control station is required (or a docking display in the cockpit).

4.2.3.2 Baseline Docking System Description. Figure 4-12 illustrates the double ring and cone concept which could be used for the docking mechanism. The docking mechanism is identical on each vehicle and consists, basically, of a ring which is mounted to the spacecraft (payload module) through four or more pairs of shock absorbers.

Guide fingers are located at intervals on the ring and project outward at an angle of approximately 45 degrees. The fingers are elements of a cone. During impact, the rings on each vehicle are guided into concentric contact by the fingers. Capture latches built into each guide finger interconnect the rings. The capture-latch geometry is designed so that at least two diametrically opposite latches must engage to prevent subsequent separation of the rings. The shock-absorbers arrangement provides a high degree of compliance so that the ring may displace both laterally and angularly

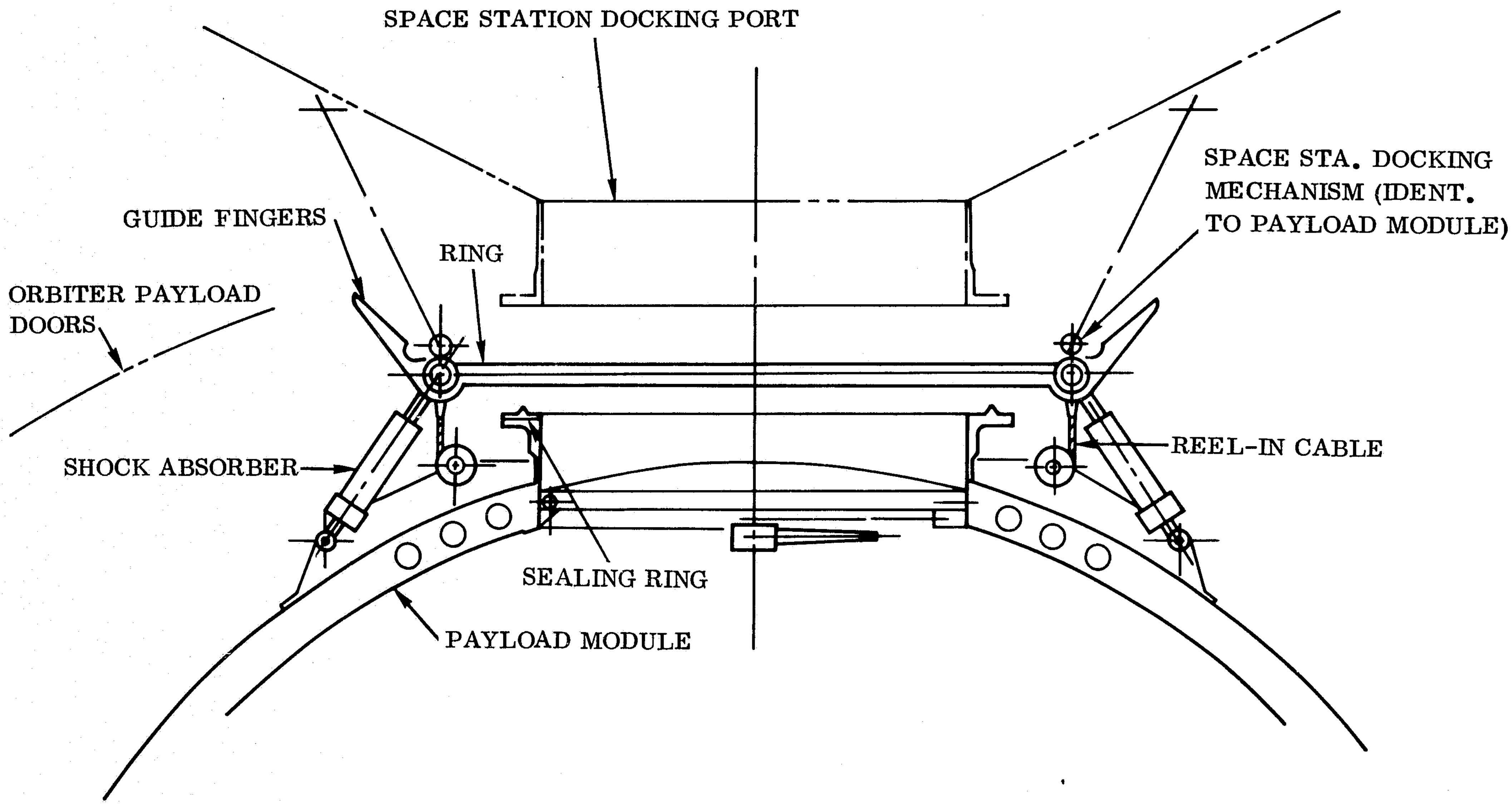


Figure 4-12. Baseline Docking System Schematic

to compensate for misalignments between the vehicles at contact. All impact loadings are attenuated through the shock absorbers with resultant energy absorption to prevent energy storage and subsequent rebound. Shock-absorber peak loads will be only a few hundred pounds because of the long strokes of the shock absorbers. A cable and winch system is a proposed type of ring-retraction mechanism which could be used to bring the vehicles into position for final structural interconnection.

In the double ring and cone concept, the crew-and-cargo transfer tunnel, the docking mechanism, and the structural interconnecting mechanism are concentric but are independent of each other.

The selection of the interface diameter (baseline = 5.0 ft) will involve tradeoffs with several parameters during the process of integrating the docking system with the space station and other space system configurations. Other variables dependent on the station configuration will be the number of shock absorbers and the number of guide fingers. The guide fingers may also be arranged to provide roll indexing, if required. Several proposed concepts for final structural latching and tunnel interface sealing exist; however, at this time, no one concept is favored above the others. A major design goal will be to minimize the need for close tolerances. The docking, structural latching, and separation functions can be performed with the system on one vehicle inactive; i.e., docking-ring retracted and, thus, immobilized. Both docking mechanisms may be active, but the performance is not necessarily improved.

4.2.3.3 Docking Operations. The delivery of personnel and cargo to the space station can be handled in a variety of ways. Several concepts under investigation are shown in Figure 4-13 and are based on the use of modules described in Section 4.2.1

Concept A shows the space shuttle personnel module docked to the space base logistics area. All cargo in modules directly behind the personnel module are transferred to the space base via the single docking port. This requires interconnecting module mating hatches. Module removal is not required with this concept.

Concept B uses a rotating adapter which allows attaching the modules at the end for a more direct cargo transfer operation than Concept A. The vehicle could hard dock at the docking port using the system described in Section 4-13 or the auxiliary docking members could be used for vehicle alignment with the space station.

In Concept C the payload modules are removed from the orbiter payload bay by (1) a mechanical manipulator in the bay, or (2) a space station manipulator. The end of the payload module is then coupled to the space station port. In this concept, a separate docking system will probably be required between the space shuttle and the station (in addition to the coupling between the module and the station).

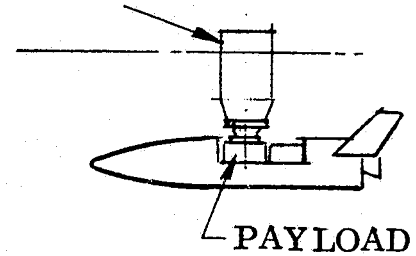
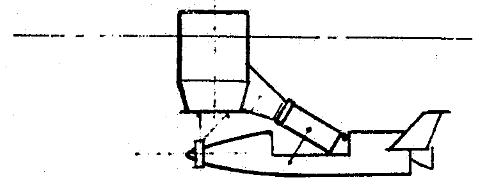
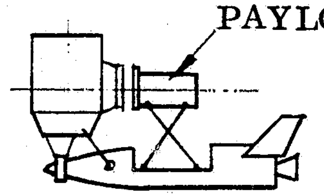
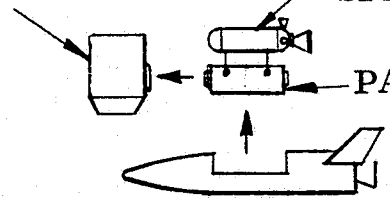
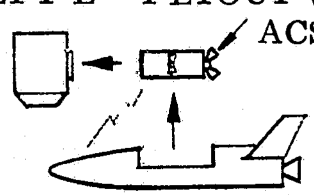
	ADVANTAGES	DISADVANTAGES
CONCEPT A - DIRECT MODULE DOCK (Not Removed) 	Direct (minimize orbital operations)	Impact of large masses on space station center of gravity and inertia
CONCEPT B - ROTATING ADAPTER 	Improved multi module cargo flow. Possible use on alternate missions	Added adapter complexity
CONCEPT C - MODULE TRANSFER (Mechanical) 	Orbiter mission time reduced. Possible use on alternate missions	Added mechanism complexity
CONCEPT D - MODULE TRANSFER (Space Tug) 	Orbiter docking not required	Added vehicle in system
CONCEPT E - FLYOUT (ACS Pack) 	Orbiter docking not required. Similar to free flying experiment module deployment	Added ACS complexity.

Figure 4-13. Orbital Space Station Docking Concepts

Concepts D and E do not require space shuttle space station docking.

Concept D uses a space tug. The tug attaches to the module, removes it from the orbiter, and finishes the operation. This method requires that a space tug be at the target or that the orbiter carries a space tug as part of its payload.

Concept E uses modules equipped with their own attitude control system. An operator in the personnel module can handle all transfers in a manner similar to the space tug. The modules will be removed from the payload bay and then flown to the station to finish the operation.

It should be noted that combinations of the above concepts are feasible; for example, Concept A for personnel and Concept E for cargo modules. Such variations have not been evaluated but appear to warrant further investigation. Selection of a concept requires complete analysis of all alternate mission requirements.

4.2.4 GROUND OPERATIONS. Space station/base dry cargo will be packaged in the modules within the logistic handling area. Upon arrival of the orbiter element at the logistic area (the element is towed in the horizontal mode) the cargo doors open and the modules are inserted by means of a loading device (monorail or overhead crane). The module is properly secured within the orbiter element, the cargo doors close, and the element is towed to the launch pad. For a dual type payload, one later requiring wet cargo (cryogenic), the dry cargo procedure is followed until the element is erected and total fueling commenced at the launch pad.

Passenger modules will be treated as dry cargo and loaded into the orbiter vehicle in the payload logistic area. The passengers will not be embarked until the vehicle is erected and the launch sequence commenced at the launch pad.

Preliminary tradeoff analyses have considered both horizontal payload insertion at a special logistic facility and on-pad payload insertion with the vehicle in the vertical position. These tradeoffs are discussed in Volume IX, Ground Turnaround Operations and Facility Requirements.

4.2.5 MULTI-MISSION CAPABILITY. The multi-mission payload capabilities of the space shuttle using the modules developed in this section are illustrated in Figure 4-14. The payload mixes shown are limited to space station logistics and related experiment modules. The passenger/cargo module shown is the baseline module defined in Section 4.2.1, and the loading conditions assume a space shuttle payload capability of 50,000 pounds.

This matrix illustrates the many module combinations available to obtain space shuttle multi-mission capability. The module loadings shown are based on maximum payload capabilities, and many other combinations are possible in off loaded conditions and/or when other missions discussed in Sections 4.3 through 4.7 are considered.

MISSION MIX	MODULE LOADING (60 FT BAY; 15 FT TYP)	LOADING CONDITIONS (MAXIMUM)
Passengers (Max)	P | P | P | P	48 Men - 43, 700 lb total
Passengers + Cargo	P | P | P | C	36 Men - 12, 900 lb available for cargo
Passengers + Cargo + Experiment Module	P | C | E (30 FT)	12 Men - 12, 000 lb cargo - 21, 700 lb available for experiment module
Cargo (Max)	C | C | C | C	26, 600 lb available for cargo
Cargo + Experiment Module	C | E (45 FT)	32, 100 lb available for experiment module 1200 lb for cargo
Experiment Module (Max)	E (60 FT)	50, 000 lb available for experiment module
Experiment Module + Passengers	P | E (45 FT)	12 Men - 39, 600 lb available for experiment module

Figure 4-14. Typical Multi-Mission Module Loadings

4.3 PROPELLANT DELIVERY

The delivery of propellant represents 44% of the missions on the NASA Nominal Space Shuttle Traffic Model. It is the most frequent mission providing primary support to the lunar program. This section discusses mission requirements, orbital operations, location of orbital operation crew, interfaces, and mission capability of the FR-1 configuration.

4.3.1 SIZING REQUIREMENTS. The mission requirements consider delivery of LO_2 and LH_2 to the orbital propellant supply facility (OPS). The propellants are used for nuclear shuttle and space tug operations. A typical lunar mission requires 200,000 lb of LH_2 for the nuclear shuttle and 68,000 lbs of LO_2 and 13,000 lb of LH_2 for two space tugs. Propellant size versus propellant weight for three delivery methods is shown on Figure 4-15. The curves are for all LH_2, all LO_2, and combined LO_2/LH_2 deliveries. Diameters of 13, 14, and 15 ft are shown to allow for tank structure and insulation as required. LH_2 payload limits are based on volume whereas LO_2 and LO_2/LH_2 limits are based on weight. For a 50,000-lb payload capability, 60 feet of length available, and transfer of 100% of the propellant to the OPS, the nuclear shuttle will require five space shuttle trips and the space tugs will require one trip each with the LO_2/LH_2 combined load. The available payload carrying capability is dependent on the tank diameter for the LH_2 delivery, but insensitive to tank diameter for the LO_2 or LO_2/LH_2 delivery.

4.3.2 ORBITAL OPERATIONS. The mission can be performed in three basic ways with regard to propellant transfer from the space shuttle to the OPS: the propellant tank can be transferred to the OPS to become part of the OPS; the propellant tank can be transferred to the OPS, be emptied immediately into the OPS and returned to the space shuttle; or the propellant only can be transferred to the OPS. These options are shown on Figures 4-16, 4-17, and 4-18.

Figure 4-16 shows delivery to an OPS facility within the context of producing maximum protection to the payload container within the shuttle bay. The 4-man personnel module configuration is shown with a 50-foot propellant tank; an alternative is the 60-foot propellant tank with the 6-man orbiter crew cab discussed in Section 4.3.3. Hard-docking to an existing unmanned OPS is required. Propellant transfer is through the line located in the stabilization boom. The use of the 4-man personnel module provides direct visibility of the transfer line hookup. In this concept, the OPS has an attitude control system which can be commanded from the orbiter or by its own preprogrammed commands. The orbiter-carried propellant tank requires its own propellant expulsion device in this concept.

An alternative configuration is shown in Figure 4-17. An unmanned OPS with attitude control capability as in Figure 4-16 is assumed in orbit. This concept, however, differs in that the orbital propellant supply tank is removed from the payload bay and attached to the OPS for propellant transfer. The orbiter can remove the empty tank shown at the top and return to Earth, or wait until propellant transfer is complete and

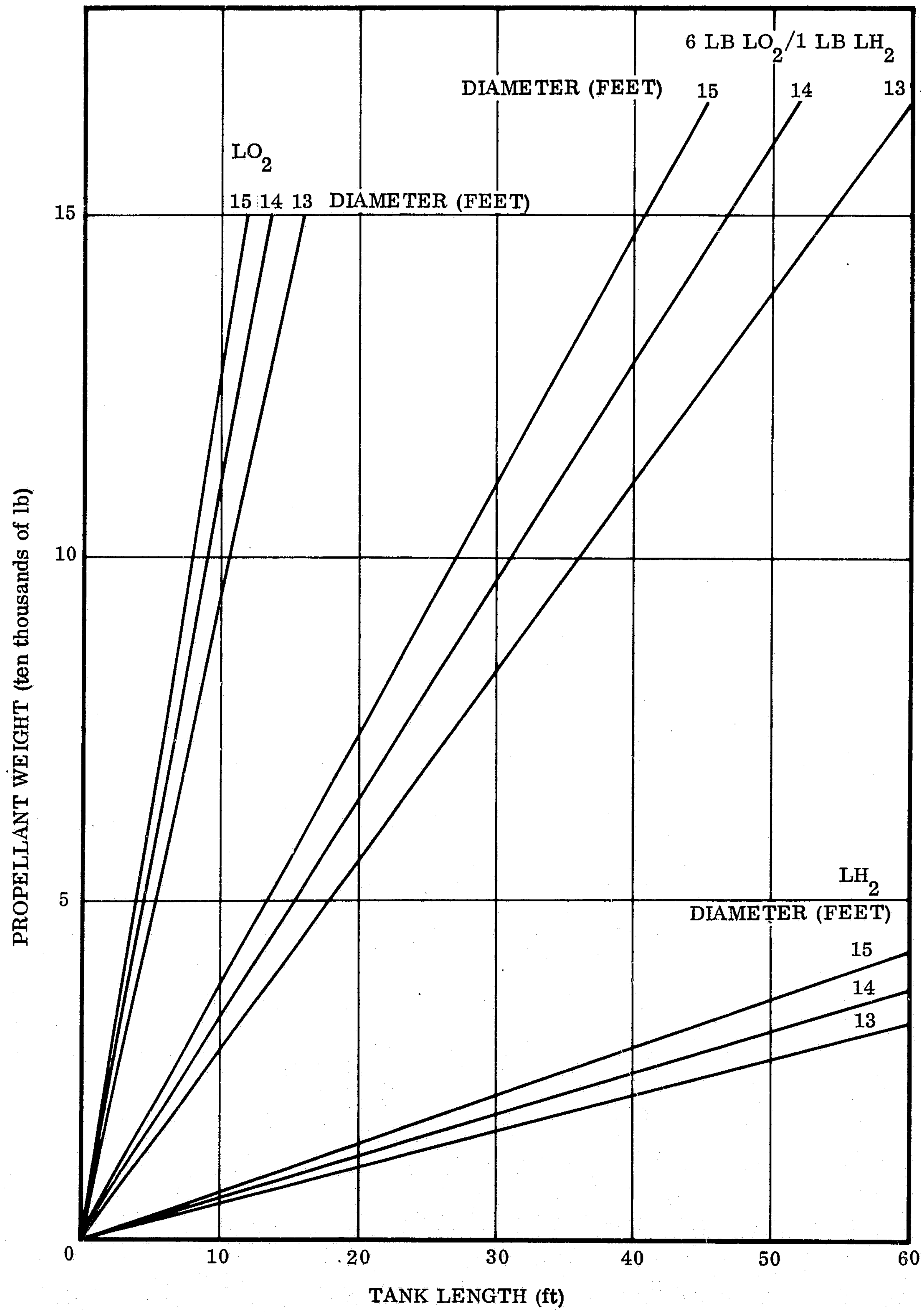


Figure 4-15. Propellant Volume/Weight Relationships

DESCRIPTION

1. OPS IS ASSEMBLED VIA LAUNCHES OTHER THAN SPACE SHUTTLE.
2. OPS HAS ACTIVE ACS, COMMANDED BY GROUND, SPACECRAFT AND OWN PROGRAMMED TOLERANCE LEVELS.
3. OPS IS NOT MANNED.
4. RESUPPLY TANK HAS OWN EXPULSION SYSTEM.
5. HARD DOCKING OF SPACE SHUTTLE.
6. STABILIZATION HOOK-UP AT BOOM BY OPERATOR.
7. TRANSFER LINE CONNECTION UNDER CONTROL OF OPERATOR.
8. RESUPPLY TANK DOES NOT LEAVE BAY.
9. THIS CONCEPT CAN HANDLE VARIOUS TANK SIZES OR MIXTURES.
10. BOOM TRANSFER LINE IS 150 FT LONG.

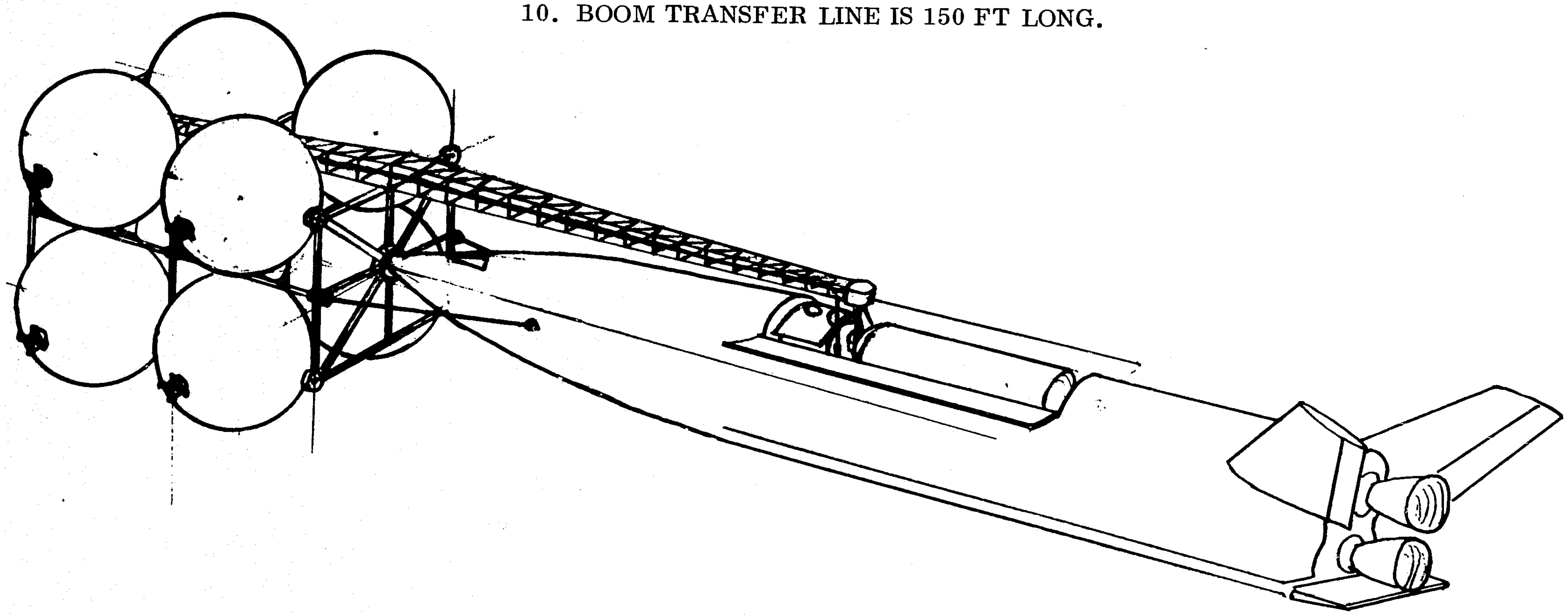


Figure 4-16. In Situ Tank, Transfer Propellant Only

DESCRIPTION

1. OPS IS ASSEMBLED VIA LAUNCHES OTHER THAN SPACE SHUTTLE.
2. OPS HAS ACTIVE ATTITUDE CONTROL, COMMANDED BY GROUND, SPACECRAFT, AND OWN PROGRAMMED TOLERANCE LEVELS.
3. OPS IS NOT MANNED.
4. RESUPPLY TANK HAS OWN EXPULSION SYSTEM.
5. NON-IMPACT DOCKING FOR TANKER. DOCKING OF CONTROL MODULE. OPS IS ESSENTIALLY PASSIVE ALTHOUGH ATTITUDE CONTROL CAN BE EXERCISED BY OPERATOR
6. 29,000 LB LH_2 PER TANK.

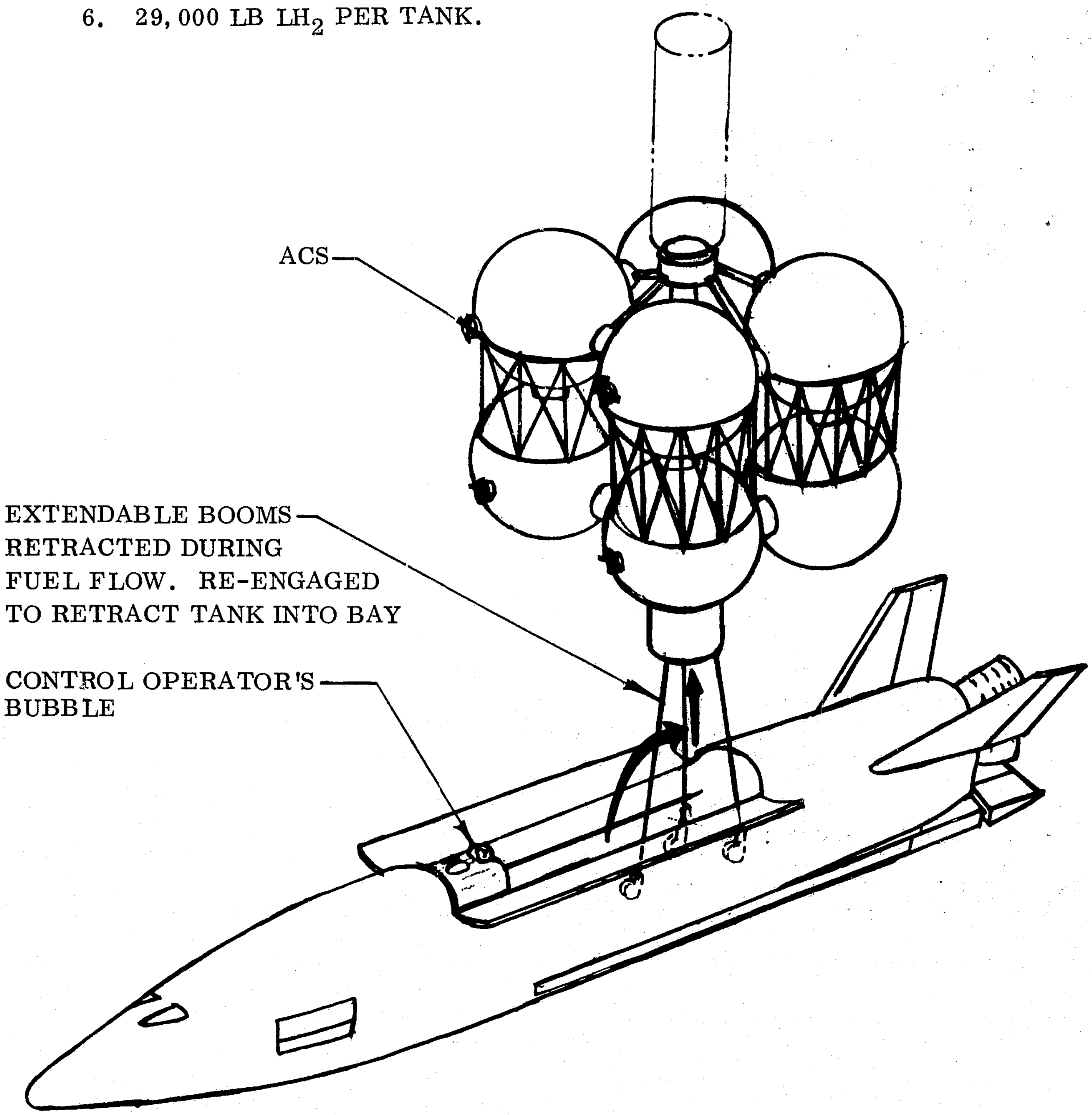


Figure 4-17. Transfer Tank

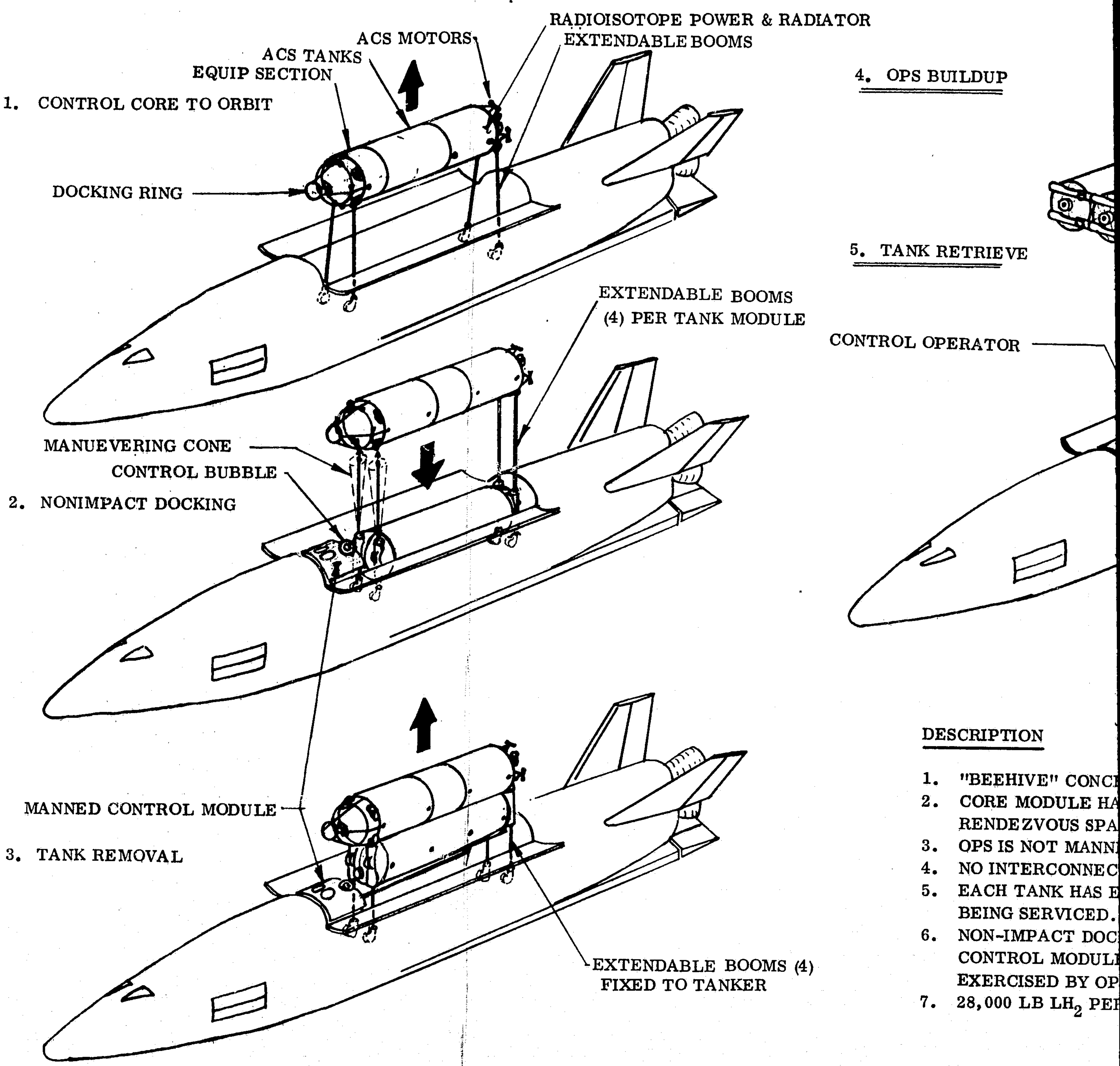
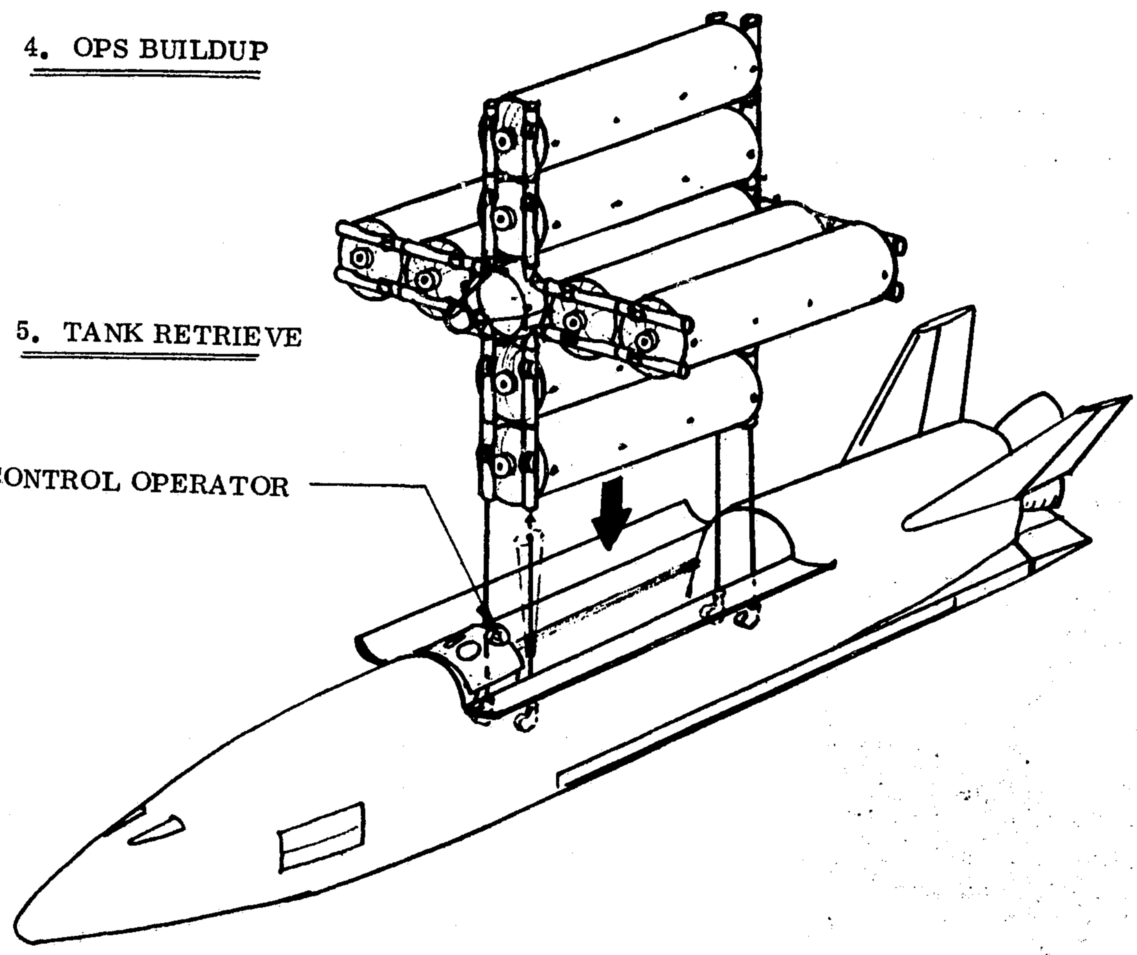


Figure 4-18. Accumulative Tanks

FOLDOUT FRAME 1

ADIATOR

LE



(4)

DESCRIPTION

1. "BEEHIVE" CONCEPT OF ACCUMULATED CELLS (TANKS).
2. CORE MODULE HAS ACTIVE ATTITUDE CONTROL, COMMANDED BY GROUND, RENDEZVOUS SPACECRAFT, AND OWN PROGRAMMED TOLERANCE LEVELS.
3. OPS IS NOT MANNED.
4. NO INTERCONNECTION OR PUMPING BETWEEN TANKS.
5. EACH TANK HAS EXPULSION SYSTEM SUBJECT TO COMMAND OF SPACECRAFT BEING SERVICED.
6. NON-IMPACT DOCKING. DOCKING UNDER CONTROL OF OPERATOR AT STATION OF CONTROL MODULE. OPS IS ESSENTIALLY PASSIVE ALTHOUGH CONTROL CAN BE EXERCISED BY OPERATOR.
7. 28,000 LB LH_2 PER TANK.

return with the same tank it brought up. The tank is shown being transferred by extendable booms under direct control of the operator in a 4-man personnel module. Following transfer of the tank, the booms can be retracted and propellant expulsion initiated. Following propellant transfer, the booms are engaged and the tank is removed from the OPS and replaced in the payload bay.

Figure 4-18 illustrates a buildup of an OPS facility using the orbiter to supply the tanks. The first delivery is a control core which contains ACS tanks and motors, power supply, and a docking ring for future servicing and for spacecraft being tanked at the OPS. The next deliveries are propellant tanks, which are attached to the control core and then removed from the orbiter. Each tank has an expulsion system subject to command of the spacecraft being serviced. Following emptying of the propellant tanks, the orbiter retrieves the tanks for refill.

In general the same problems exist in varying degrees for fluid transfer in orbit as for fluid transfer on Earth, such as:

a. Explosive, toxic, and chemical hazards.

b. Childown and external heat transfer of lines, tanks, and other hardware where cryogenic fluids are involved.

c. Fluid contamination, such as from the absorption of pressurant gas.

d. Loss of fluids due to boiloff, leakage, and general inefficiency of the transfer process.

e. Tradeoff between fluid transfer through lines and fluid transfer by tank replacement or other intermediate steps.

f. Determination of the most efficient hardware and expulsion methods.

The effects of transfer in orbit on these problems, as compared to normal Earth conditions, are:

a. Absence of natural one-g force field. The basic problem arising from this fact is that special methods are required to maintain gas/liquid separation, orientation and/or acquisition of liquid for transfer. Low-g fluid statics, dynamics and heat transfer are not completely understood and test data are very limited. One advantage, however, is that items are essentially "weightless" and large masses can be physically moved short distances with relatively little effort.

b. Vacuum environment. The main disadvantages are in sealing requirements, effect of high vacuum on materials, and the lack of protection from radiation and micrometeorites. One advantage is that explosive, toxic and chemical hazards are reduced since there is no oxidizer naturally present and shock waves are not transmitted. Also, escaped fluid pressures are so low and dispersion so complete that

many potential hazards are non-existent; e.g., unless confined, any gas will completely disperse at zero pressure since the mean free path of the molecules becomes essentially infinite. In addition, fouling of instrumentation due to condensation of water vapor from the surroundings will not be a problem.

c. Transfer hardware weights are very important due to the cost of putting a pound into space. Brute force methods and equipment would not be as practical as on Earth.

d. The use of personnel to perform transfer functions is somewhat more limited in orbit than on Earth.

Table 4-7 presents a summary of the areas to be discussed in more detail herein. A rapid comparison is made of the three transfer techniques shown on the previous figures. The following subsections cover propellant orientation and transfer, and transfer efficiency.

Table 4-7. Concept Comparison

	Transfer Propellant Only	Transfer Tank, Empty, Return to Orbiter	Transfer Tank Which Becomes Part of OPS
Fluid Orientation - For Transfer to OPS	Linear acceleration Rotational acceleration Fluid rotation Mechanical-Bladder	Not required	Not required
Propellant to OPS	Pump/Pressure	Pump/Pressure	Mechanical
Transfer Efficiency to OPS	<100%	<100%	100%
Chilldown of Transfer Lines	Yes	Yes	Not required
OPS Venting Required During Transfer	Possibly	Possibly	
Cooldown of Receiver Tank Required	Possibly	Possibly	No
Weight of Orbiter Delivered Tank Insulation	Lightest		Heaviest
Expulsion Equipment for Propellant	Part of Payload	Part of Payload	On OPS

4.3.2.1 Propellant Orientation. For those concepts requiring transfer of propellant only, a primary problem is to obtain the fluid for transfer in a single state, since in the absence of applied settling forces, the liquid and gas can be mixed within the tank. Normal vehicle disturbing forces from attitude control, drag, and gravity gradient will be present but are not necessarily conductive to positioning the fluid where transfer can be easily accomplished. The use of linear acceleration, tank rotation, fluid rotation, and positive containment will be discussed.

Linear acceleration consists simply of applying an accelerating force along the axis of the propellant tank in order to position the liquid at one end of the tank. In this method of transfer both the OPS and orbiter must be accelerated. The main disadvantage of this method is that orbital perturbations are created during tanking unless special orientation is accomplished. The orbital perturbation can be minimized by thrusting continuously perpendicular to the original orbit plane for one orbital period while transferring propellants.

For other than orbiting transfer the displacement from nominal path through increase or decrease in velocity as it affects the overall mission must be considered. This could be an advantage for some missions where transfer could be accomplished during a normal mission acceleration period.

Quite a bit of study has been done on the operational aspects of the linear acceleration system under NASA contracts. The main advantage of this system is in its similarity to orientation as encountered on Earth, allowing for conventional vehicle designs and a fairly high state of the art in transfer procedures and equipment.

Given enough acceleration, the method is bound to work; however, it will probably be desirable to keep the acceleration as low as possible in order to minimize orbital perturbation and expended propellants. The main developmental problem is in determining the minimum acceleration and time required to satisfactorily settle the propellants and ensure a minimum of trapped gas in the liquid outflow and/or the transfer efficiency as a function of acceleration.

In the rotational acceleration method, the propellant tank is rotated about its longitudinal axis to cause liquid to exist around the side of the tank. Both OPS and supply or only the supply may be rotated. Where only the supply tank is rotated, a slip or rotating transfer joint is required. Such joints are sometimes used with fluid test set-ups on ground-based centrifuge machines and could probably be adapted to space use.

One of the main considerations with this system is the time it takes to get the fluid rotating. If time is an important factor baffles can be installed to decrease this time, but this will add weight. Also, residuals can be significant unless special side traps are used.

This method has low fuel requirements (once spin-up is obtained no further acceleration is needed) and minimum orbital or trajectory perturbations.

Fluid rotation is based on the same physical phenomenon as tank rotation except only the fluid itself is rotated, thus eliminating the need for rotating the OPS or providing for a rotating transfer connection. Three methods of providing the necessary vapor/liquid separating vortex are by using mixers, pumps, or vortex type separators.

Positive containment methods, as defined herein, include all items such as bladders, bellows, pistons. With these methods, the liquid to be transferred is separated from the ullage gas by a positive physical barrier. The use of such a device is straightforward for storable fluids. In using one with cryogenics, boiling can occur within the positive expulsion device and fluid orientation within the device may be unknown. At initiation of transfer, however, the ullage pressure can be raised above the trapped fluid vapor pressure such that essentially all trapped vapor is condensed. Mixing of the trapped fluid may be required to assure complete condensation and prevent excessive stratification. It should be noted that condensation of trapped vapor may not be necessary for transfer, depending on the required efficiency and the expulsion method used.

4.3.2.2 Propellant Transfer. Propellant transfer can be handled by expulsion methods or mechanical transfer of the tank itself to the OPS. The types of expulsion methods which can practically be considered for a particular application depend in part on the type of liquid acquisition or containment device being considered. The use of most pumping and pressure-feed methods are applicable to essentially all the liquid containment methods of Section 4.3.2.1. There are, however, limitations on the type of pressurants and pressurant temperatures when the expulsion gas is to come into contact with the propellant as opposed to cases where bladders, diaphragms, bellows, or similar positive separation devices are used. The various expulsion methods are discussed in the following paragraphs.

The following types of pumps can be considered for the space transfer application:

a. Centrifugal

b. Axial

c. Mixed flow

d. Positive displacement

e. Jet pumps

Most of the pump work which has been done which is applicable to fluid transfer in space has been done in connection with feeding propellants to vehicle engines including chilldown and recirculation pumps. Some consideration should be given to the possibility of using such a pump already present on the receiver or supply vehicle to also perform the transfer function. The main difference in the transfer application is that

pressure rise requirements will in general be significantly lower than for the engine application. This allows the consideration of axial flow and all-inducer types of pumps. The main engine pump stages for present day space and booster vehicles are of the centrifugal type, sometimes with an inlet high specific speed inducer section designed to operate at close to inlet saturation pressures (low NPSH). Operation with low inlet pressures is important to the minimizing of pressurant requirements.

The use of a gas or vapor to expel fluid from a storage tank is a fairly well established technology. Methods used for engine feed will also be applicable to the space transfer of fluids with, however, the increased potential of gas blow-through during the low-g draining. Analytical studies have shown this to be a potential problem.

Another potential problem is that a sudden tank pressure reduction can occur at initiation of pressurization at low-g. This is caused by the injected gas inducing liquid spray into the ullage space and cooling the ullage gas. The gas inlet momentum must be kept at a minimum. For this purpose the Centaur has a 10 1/2 in. diameter screened inlet. This is especially critical where pressurant is used to provide pump NPSH. A sudden reduction in pressure may also be harmful to the structural rigidity of the tank.

Two areas of interest in the application of expulsion techniques are the required interface between the propellant tank and the OPS and OPS receiver tank venting versus non-venting operation.

The OPS interface is defined as the piping and hardware connecting the receiver tank to the supply through which the fluid is transferred. Connection can be either flexible or rigid. Basic problems are in general the same as in ground transfer, except hardware weights, reliability, and remote operation are more critical. The OPS interface needs reliable leak-tight coupling connections and shutoff valves. A shutoff function can be built into the coupling, but one or two additional valves would probably be needed to control flow while connected.

Where cryogenic fluids are involved, cooldown of hardware mass and insulation requirements are important considerations.

Tanker studies which have been performed indicate a rigid connection as being the best method. The main reasons are:

a. Automatic hookup is more feasible.

b. Minimum attitude control sequencing is required.

c. Propellant transfer lines ar not subject to structural stress.

d. Would work better where transfer uses linear acceleration.

Further consideration should, however, be given to the flexible or non-rigid type of connection since the need for final docking of the two vehicles would then not be required.

The mechanics of coupling would be similar to that used for airplane refueling. Coupling make and break latching and valve mechanisms could be similar. The method of boom extension and guidance could also be similar, only simpler, since aerodynamic perturbations are absent. Extensible metal tape booms, which could be used to extend a line to another vehicle, are useful for space application.

The design of the OPS must consider venting or non-venting during fill with a tradeoff between tank pressure rise (structural weight penalty) and complexity.

If venting is required, a method for venting only gas will probably be needed. Even where acceleration is applied to both receiver and supply it has been calculated that instabilities of the liquid surface during filling can be significant. The sloshing of fluid entering the tank can be reduced by providing proper baffling. Assuming the possibility of liquid occurring at the vent inlet and a requirement to vent only gas, a vapor/liquid separator would be needed.

Cooldown of the receiver tank is an important consideration in determining allowable fill rates, venting rate requirements, and pressure levels expected, especially where cryogens are involved. Overpressure due to excessive vaporization rates during fill will be the primary consideration; however, an initial reduction in pressure can occur due to heat transfer from the ullage gas to liquid spray within the tank. This should not be a significant problem, but could cause a loss of structural rigidity if the pressure were reduced far enough. Such a reduction in pressure cannot occur when significant amounts of liquid spray are kept from the ullage.

4.3.2.3 Transfer Efficiency. Transfer efficiency is defined, herein, as the ratio of the amount of liquid transferred to the original amount stored. The overall transfer efficiency is not really a technology in itself, but is a function of the type and efficiency of individual items discussed in Sections 4.3.2.1 and 4.3.2.2. The following factors contribute to efficiency or inefficiency of the overall transfer process.

a. Leakage Through Joints and Interconnections. Most hardware can contribute to this problem. Vacuum and space environment can have a detrimental effect on static and dynamic seals.

b. Venting. During storage due to heat leaks from insulation, supports, tank inlet and outlet lines, and internal power generation. Chilldown of transfer line and receiver tank also causes propellant vaporization.

c. Fluid Residuals. Dependent upon positioning and/or pumping methods used. Primary consideration is vapor pull-through at low-g; i.e., the point at which sufficient vapor enters the transfer line such that pump cavitation can occur (if applicable) or that no more liquid can be transferred. The remaining liquid represents residuals.

Pull-through and blow-through for a given expulsion and orientation configuration depends on transfer rate and flow velocities. It may be advantageous to reduce flow rates toward the end of transfer.

The transfer efficiency effect on mission requirements is shown on Figure 4-19. With 100% efficiency and an assumed 45,000-lb delivery capability the basic lunar mission requires 4.7 flights of LH_2 and 1.5 flights of LO_2. LO_2 transfer efficiency can drop to 76% before more than two flights are required. LH_2 transfer efficiency can drop to only 96% before more than five flights are required. The effect of transferring from the orbiter to OPS and OPS to the nuclear shuttle and space tugs is shown by assuming identical efficiency for each transfer. The baseline mission for 80% efficiency and two transfers would require over 10 flights. This compares to only eight flights if the tanks are mechanically transferred and only one propellant expulsion operation takes place. It is therefore recommended that mechanical transfer of the tank be considered at this time as the primary candidate for the OPS operation.

4.3.3 LOCATION OF ORBITAL OPERATIONS CREW. The desirability of locating the orbital operations crew forward with the orbiter crew is best illustrated by Figure 4-20. The baseline number of flights to support a lunar mission, 2 LO_2 deliveries and 5 LH_2 deliveries, would increase by one if a 10-foot personnel module were included in the payload bay. Figures 4-16 through 4-18 all illustrate transfer concepts using a 2 to 4 man personnel module. Figure 4-21 illustrates a variation of the tank transfer method. The OPS is different, but more important is the small aft docking control station shown in the payload bay. This is reached from the orbiter crew cab by the transfer tunnel and is positioned to allow operator visibility without reducing payload volume. The small size of this control station is possible due to its use only during the transfer operation.

The decision on location of the operations crew should be based on a more detailed analysis of the selected transfer operation and the OPS orbiter interface. At this point it seems that transfer of the propellant module using the small aft-control station is the most desirable technique.

4.3.4 INTERFACES. This section will deal primarily with the orbiter, propellant module, and launch pad interfaces.

Two basic interface concepts have been investigated for the removable propellant module concept. The first fully integrates the propellant module servicing lines with the orbiter propellant system, while the second system has completely independent service systems. There are other concepts which are combinations of these two, but for relative comparison purposes these two types of systems have been evaluated.

The separated system locates the internal disconnect panel aft of the propellant module. This location provides space envelopes for panel assembly, flex line sections, activation mechanisms, and structural supports. The overboard umbilical connections,

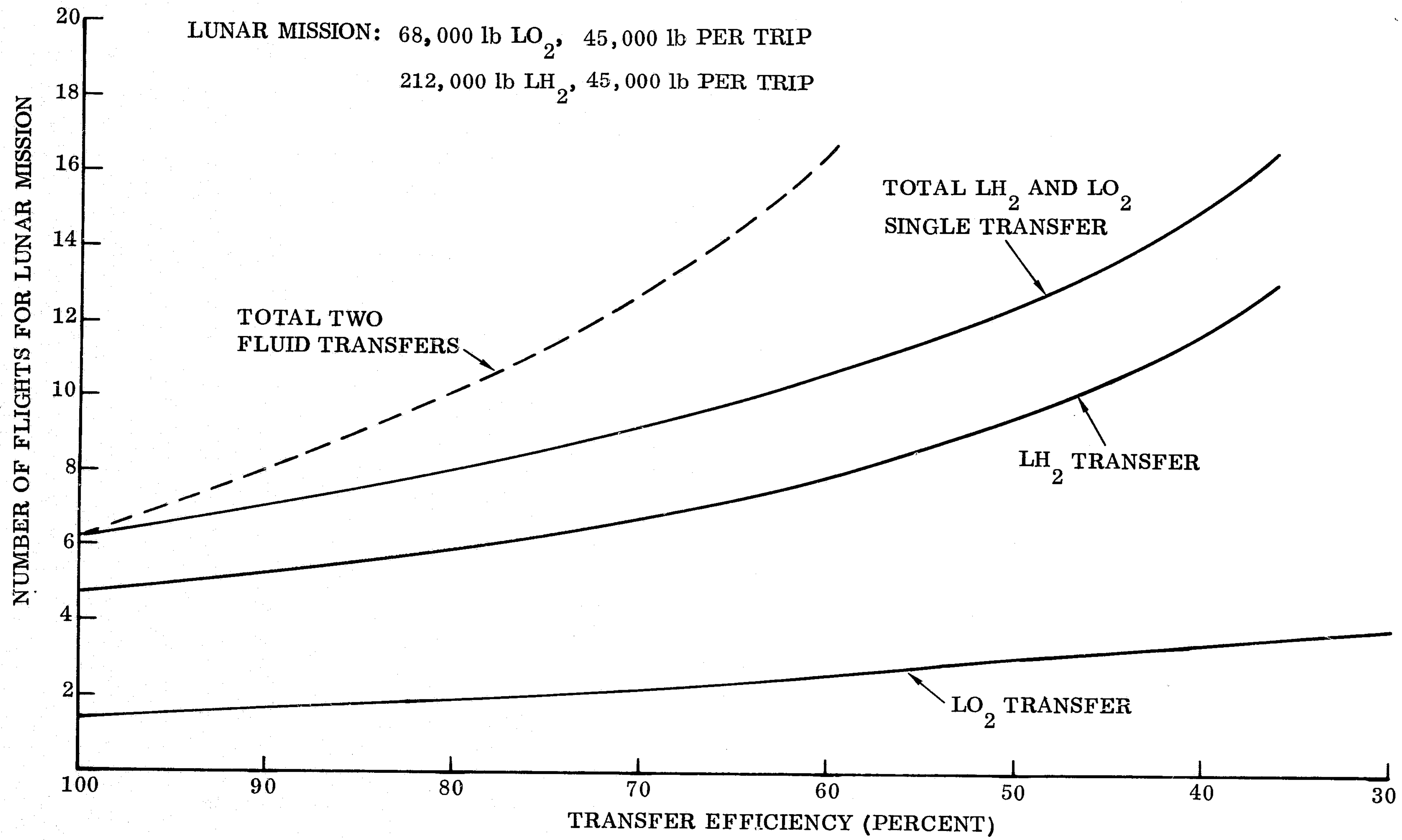


Figure 4-19. Propellant Transfer Efficiency

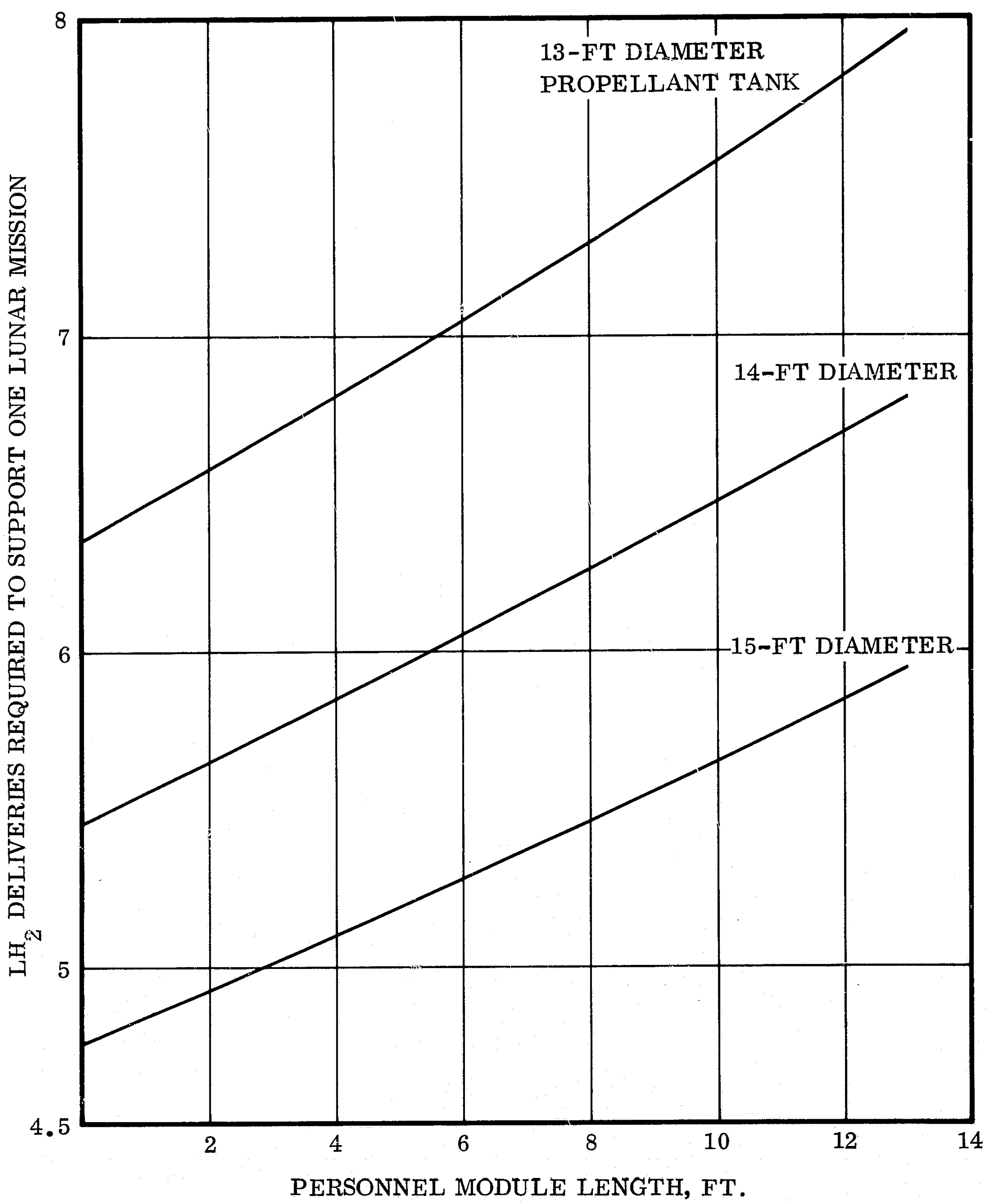


Figure 4-20. Personnel Module Length vs. Required Orbital Flight for LH_2 Delivery

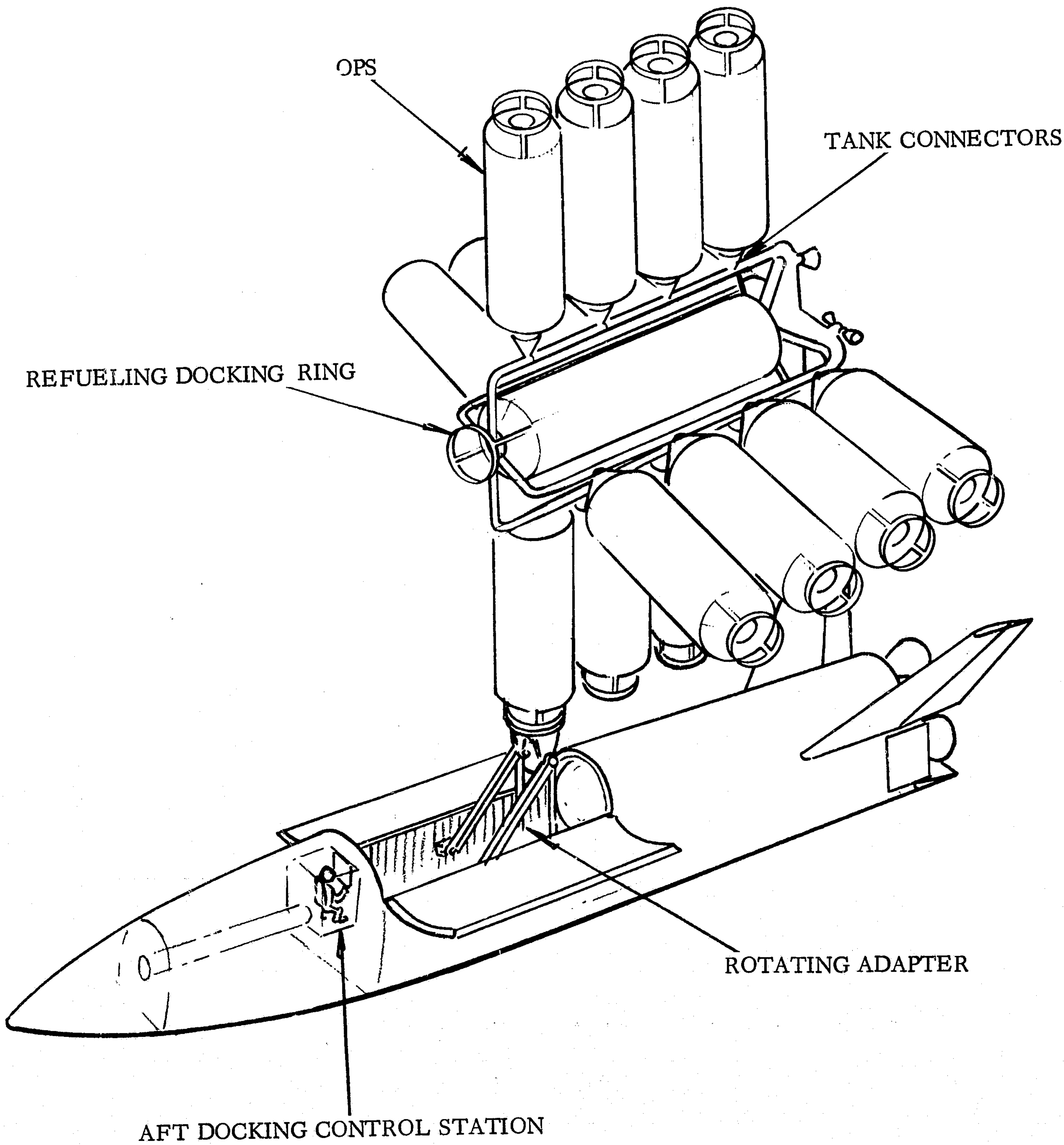


Figure 4-21. Tank Transfer Concept

however, are mounted on a separate panel which may require additional weight and ground actuation systems. Tradeoffs exist when considering common or separate overboard umibilical panels. In some cases, the vehicles may be penalized with additional line lengths to accomplish common panels. Efforts should be made to place the burdens on the ground systems rather than on the vehicles.

Fill and drain shutoff valves are located near the propellant module. The fittings at the internal disconnect panel are also equipped with shutoff devices for preventing spillage. When the disconnects and shutoff valves are activated, propellants are trapped in the duct-section routing between the fill and drain valve and the panel. This requires a relief circuit, which is accomplished by routing a relief tube equipped with a check valve between the duct and the propellant module.

The disadvantages of the separate plumbing arrangements are duplication of line routings, increased number of ground umbilicals, and increased weight. Emergency dumping will use the fill-and-drain line ground umbilical connection. Because this connection will close after umbilical disconnect, a mechanical means will have to be provided to open the valve for in-flight jettisoning.

No spillage is allowed when the internal panel is actuated; therefore, each disconnect fitting is equipped with dual unsymmetrical poppets which are actuated prior to panel separation. The small reservoirs of fluid trapped between the poppet closures are purged with helium gas prior to disengagement. This is accomplished by routing a helium purge tube to the orbiter side of each disconnect. The helium requirements are small and may be supplied through the propellant ground umbilical or from the orbiter system.

A low pressure, ambient helium prelaunch purge is provided through the umbilical disconnect. The purge conditions the propellant module insulation by removing virtually all air and moisture within the shroud interior. Purge helium is fed through a manifold of external lines through the insulation blanket layers and out the seams.

The vent system must provide the capability for ground venting of propellants as well as in-flight venting.

The ground vent is routed directly to the ground umbilical island. A mechanical means will have to be provided to allow venting after launch if tank lockup is not possible during this storage period. This will be dependent on the duration and cargo bay thermal environment.

The internal disconnect panel between the propellant module and the orbiter must disconnect with no leakage, accommodate line pressures, and absorb misalignments due to temperature gradients and tolerances. The assembly must also be capable of reconnecting in emergencies with no spillage. Several design approaches involving multipoint linkages, hinged panels, track roller combination, and a telescopic beam are

available for the retracting system. Actuation can be accomplished pyrotechnically or electromechanically. This is a major area which should be exposed to tradeoff studies before a final selection is made. The integrated system design must provide for separation from and reloading into the orbiter, jettisoning propellant in case of an emergency, and ground vent capability.

The integrated installation conveys propellants from the ground umbilical, through an internal disconnect panel, through the orbiter, to the propellant module. The propellant module fill-and-drain system and vent system is interconnected with the orbiter circuits. For emergency conditions, the propellant module propellants may be dumped into the orbiter and subsequently burned through the main engines. The vent ducts are routed through the orbiter circuits to the ground umbilical panel. The basic concepts of the separated system disconnect panel and purge requirements are similar to that of the separate system.

Structural attachments between the propellant module and the orbiter are required which adequately transmit loads between the two while still allowing for deployment and retrieval of the propellant module.

The structural members available to carry the propellant module loads are two longerons, each running the length of the cargo bay along each side. Thus, the distributed axial load about the 47-ft circumference of the propellant module must be transferred to two points in the orbiter. This requires the addition of special structure to transfer the loads which may be included either as a part of the propellant module or a part of the orbiter vehicle.

A method of adding the special structure to the cargo bay consists of a distributed load ring which mates with the aft adapter ring of the orbiter and a set of oblique skin stringer frame cone segments to transfer this distributed load to two axial load pin sockets located approximately 6 ft aft of the ring. This configuration is chosen for maximum structural efficiency (minimum weight) and is about 75 percent effective in transmitting the two point loads to a fully distributed loading on the adapter ring. Latches are included on the mating ring of the orbiter to engage and hold the propellant module during orbiter maneuvering, entry, and landing. These structural latches must be reusable for multiple deployment and recovery of the propellant module. The latches are normally open and pneumatically actuated to the holding position. This provides a fail-open mode so that the propellant module can still be deployed or recovered should one or more latches become inoperable.

The load transfer between the orbiter and propellant module is accomplished by axial and lateral load pins. These load pins are supported from the orbiter longeron structure on each side of the cargo bay and are actuated by either electric motor-driven ball-screws or pneumatic cylinders. The actuators must retract the aft lateral pins far enough to allow unrestricted movement of the propellant module deployment linkage.

The axial pins mate with the sockets located on the propellant module and are loaded in single shear. The lateral pins provide resistance to vertical and side loads. These pins provide complete stability for the tank in the cargo bay during orbiter launch, orbital maneuvering, entry, and landing.

A method of extracting the propellant module from the cargo bay of the orbiter while in orbit is required. Many mechanisms are capable of performing this separation. Among those considered were slider, rotating arm, and four-bar linkage mechanisms. Figures 4-18 and 4-21 show typical linkages. Since these linkages operate only in a zero-g environment, the only forces applied are those due to inertial loads. No specific deployment time has been specified, but a gradual egress taking approximately 60 seconds will result in very low loads and linkage members with relatively small cross-sections.

A proposed NASA universal docking ring is desirable for capture and mating of the orbiter and empty propellant modules. This docking ring concept allows either identical half of the device to be active while the other half is passive.

4.3.5 MISSION CAPABILITY. The ranges of launch azimuths and orbit altitudes for this mission are shown in Figure 4-22 and illustrate the payload capability. The all LH_2 delivery is volume limited at less than 45,000 lb. The LO_2 or LH_2/LO_2 deliveries will be weight limited based on the capability shown in this figure.

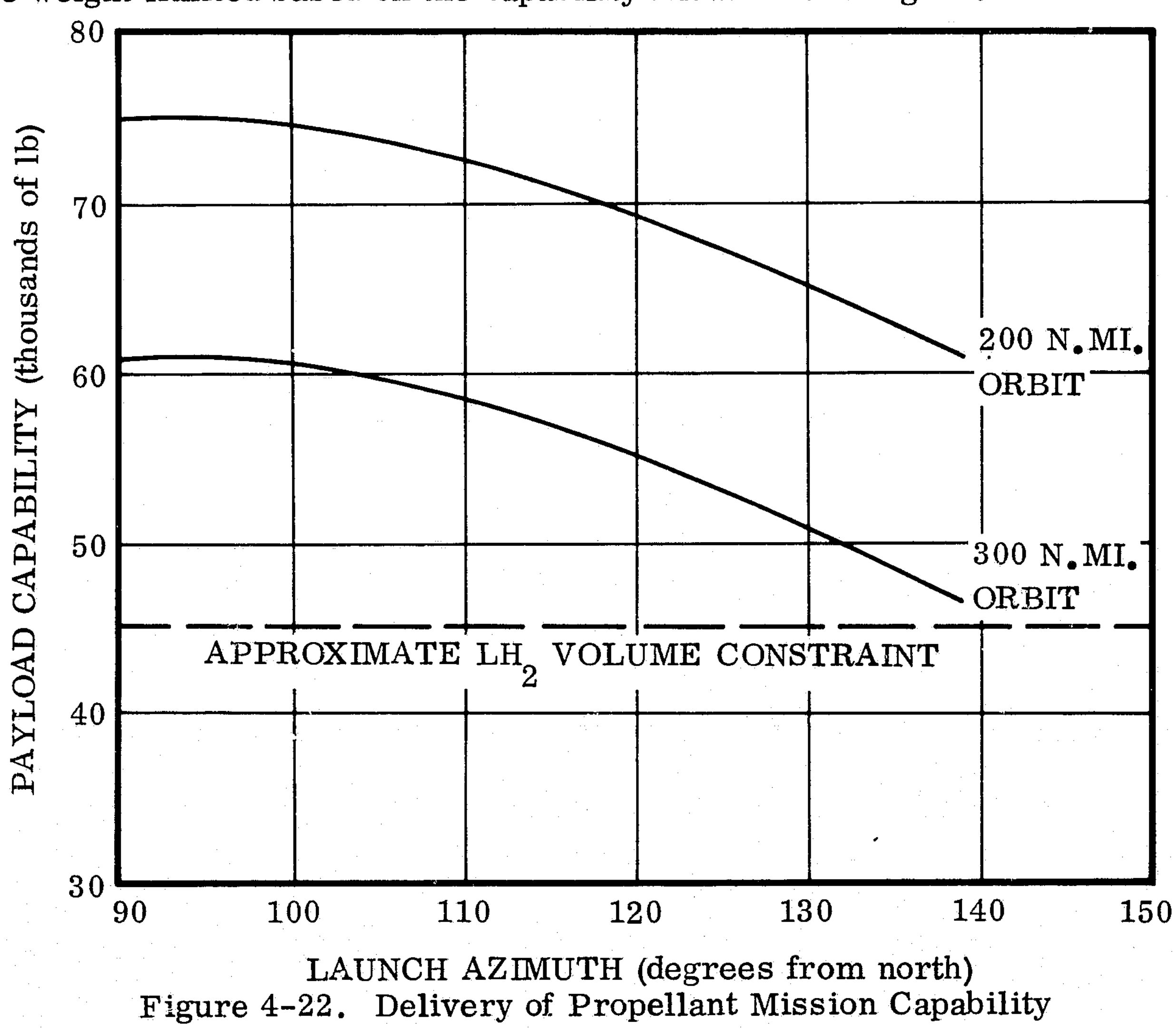


Figure 4-22. Delivery of Propellant Mission Capability

4.4 PROPULSIVE STAGE PAYLOADS

This section describes payloads that are additional propulsive stages carried in the payload bay to increase the mission capability of the basic space shuttle. The propulsive stage is designed to perform a wide variety of missions after being injected into a 100-n.mi. orbit by the space shuttle. Since it is initially injected into orbit in the cargo bay of the space shuttle and, in the reusable mode, returns to the cargo bay after completing its mission, there are unique structural and fluid interface requirements between the two vehicles which are defined in the following subsections.

Two point designs have been conducted using LO_2/LH_2 as the propellant combinations. One design is sized for an ETR launch and the second is sized for a WTR launch. The stage sizes chosen are representative of sizes required to accomplish the placement of a minimum of 5,000 lb of payload at synchronous altitude and return of the propulsion system to 100 n.mi. altitude for rendezvous with the Earth-orbit shuttle. The purpose of the design analysis was to identify areas where the unique launch conditions of this system result in design concepts different from standard space launch vehicles. The engine system used for this study is the Pratt and Whitney RL10A3-3A, which provides an excellent baseline since the operating characteristics of this engine are well known.

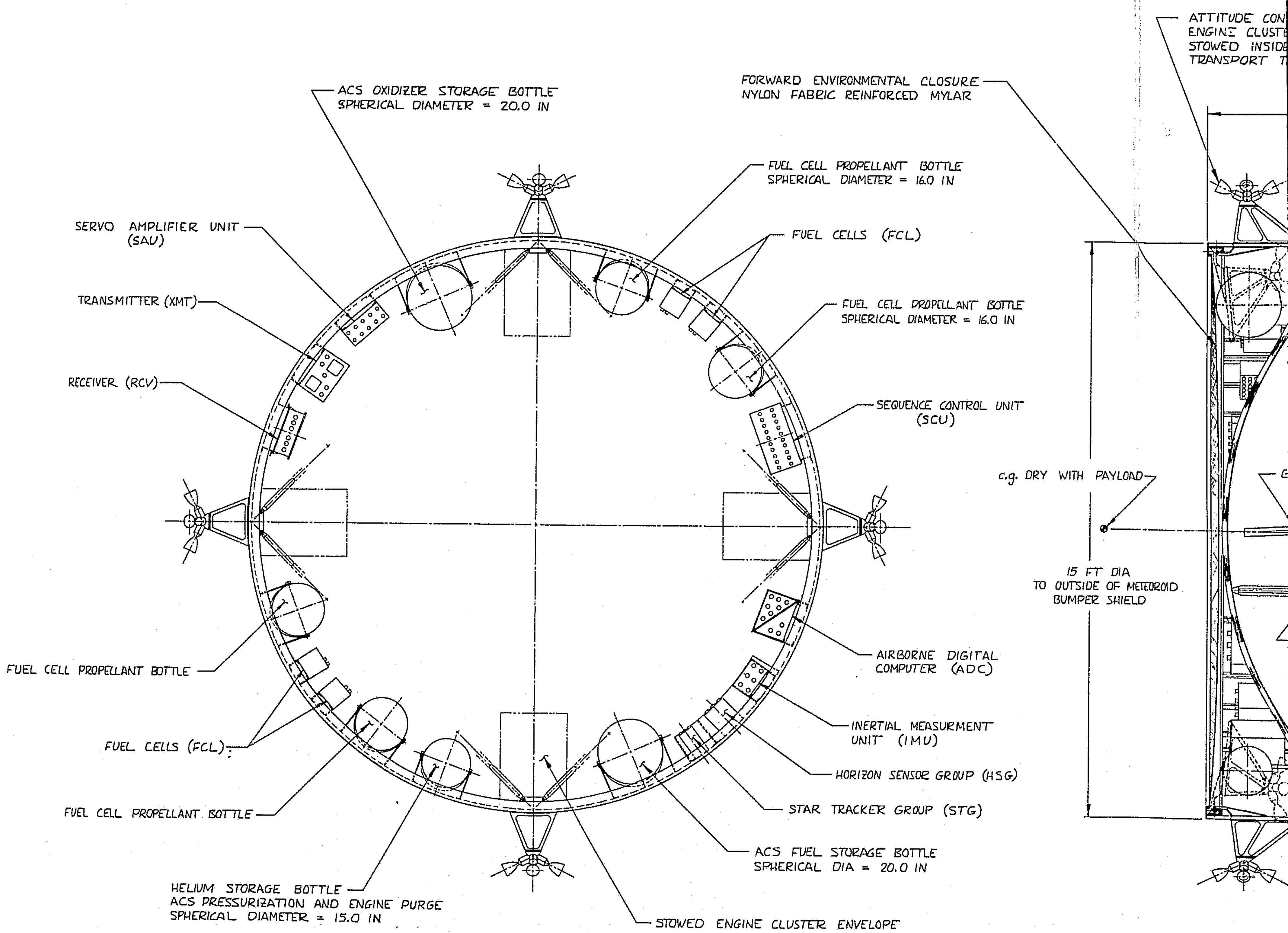
4.4.1 STAGE CONFIGURATION - WTR LAUNCH. The vehicle design is shown in Figure 4-23. The primary objective of the design study was to achieve a stage of minimum weight and length consistent with the mission requirements and Earth-orbit shuttle cargo bay volume.

Preliminary sizing studies indicate the stage size required to lift a minimum of 5,000 lb of payload from 100 n.mi. to synchronous altitude is approximately 39,000 lb (32,500 lb LO_2 and 6,500 lb LH_2). This represents a stage sized for a WTR launch capable of placing over 5,000 lb of payload into a synchronous inclined orbit from either WTR or ETR, with a return of the propulsion system to 100 n.mi.

Since the maximum vehicle diameter is limited to 15 ft, the propellant tanks were constrained to a diameter of 14 ft, allowing only six inches for super-insulation, structural support, and meteoroid bumper shield.

4.4.1.1 Propellant Tanks. In-line positioning of the two propellant tanks and the engine creates a long-stage length resulting in a greatly reduced cargo bay length available for payload. Solution to this problem was found by locating the engine aft of the hydrogen tank and surrounding it with toroidal oxygen tanks. This results in an extremely efficient stage geometry.

A 14-ft diameter LH_2 cylindrical tank with 1.38 elliptical bulkheads was chosen. This bulkhead ratio is the maximum value which can be used without the possibility of creating compression zones in the bulkhead. The hydrogen tank was sized with 5 percent excess volume to allow for ullage, hydrogen boiloff and residuals, zero-g vent valve,



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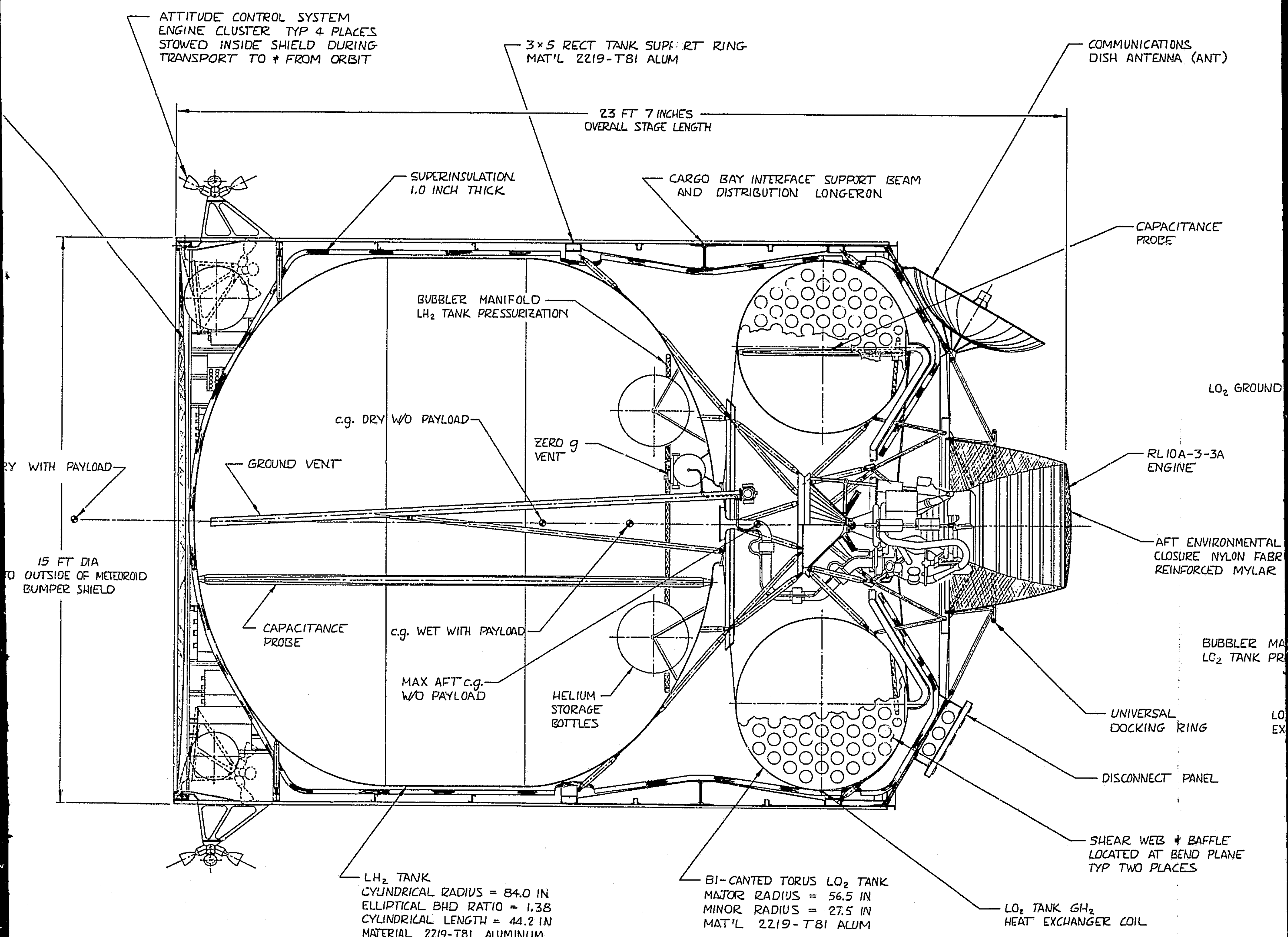

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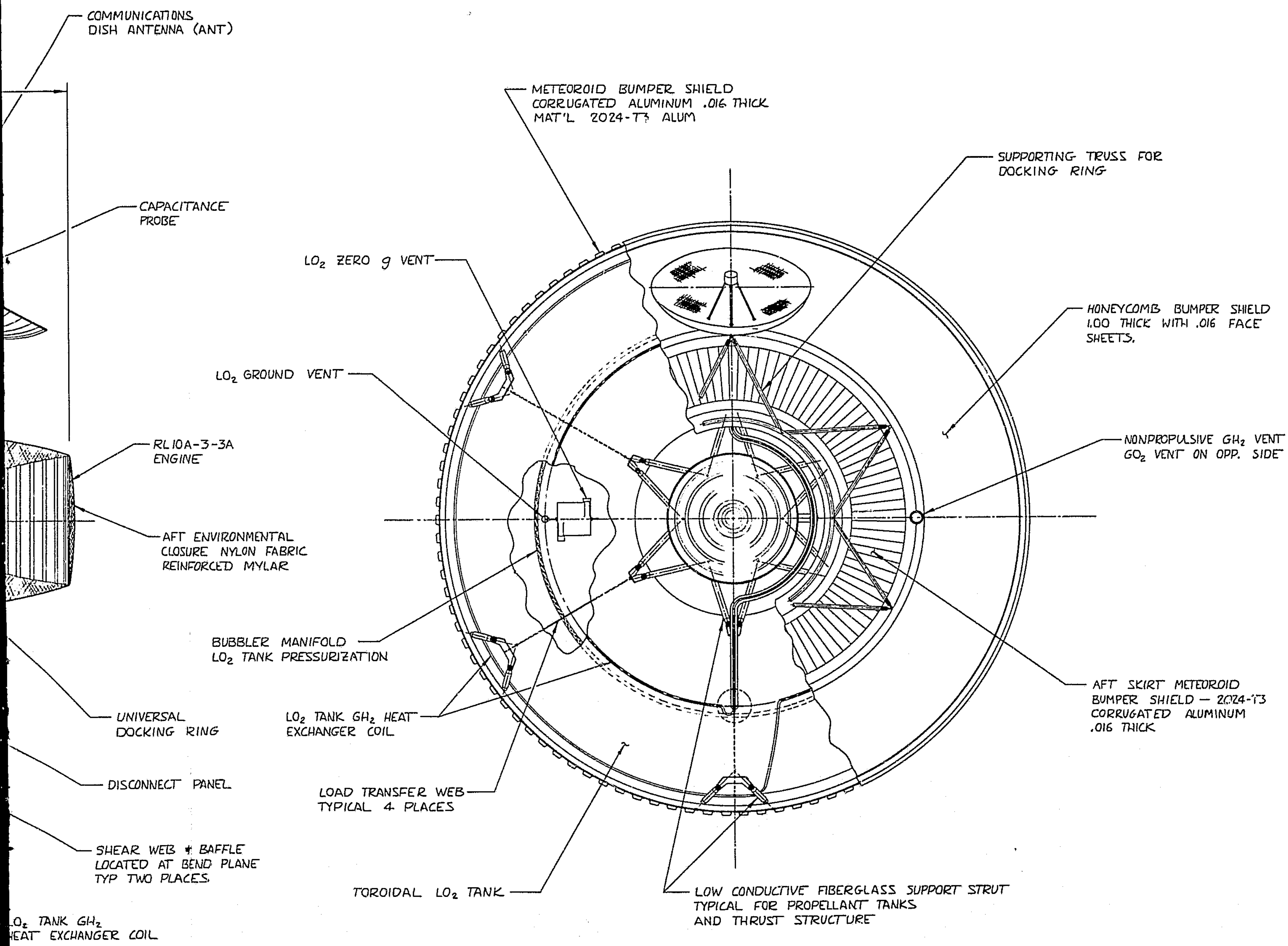

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Figure 4-23. Propulsive Stage, 39,000 lb Propellant

and helium storage bottles. The total tank volume is 1,606 ft^3 and holds 6,500 lb of liquid hydrogen. The cylindrical section of the tank is 44.2 in. in length. The tank wall thickness was determined with 27.0 psia tank pressure and a full 1-g liquid head. A greater head pressure occurs during space shuttle boost of 4g, but the propulsive stage tankage can be maintained at a considerably lower pressure during this phase of the mission. The allowable design stress used is 1.0 times the ambient yield of 2219-T81 aluminum, 44,000 psi. This results in an elliptical bulkhead crown thickness of 0.0362 in. and a cylindrical section thickness of 0.0525 in.

Toroidal propellant tanks have been proposed for use on many conceptual designs, but their actual use at this time has been very limited. Propellant residuals are held to an acceptable level by canting the torus. A simple tilt will result in excessive c.g. lateral off-sets at low propellant levels, but a flat V or bi-canted torus eliminates this problem. A bi-canted torus employs two propellant feed lines, one at each low point.

The tank is fabricated by spin forming the top and bottom halves of the torus and butt welding them together. Canting is accomplished by cutting a small wedge out of opposing sides of the torus, inserting a shear web, and butt welding the two torus halves back together.

4.4.1.2 Structural Supports. The structural support for the propellant tanks and engine is provided by trusses and vee truss segments. The engine is mounted to a small titanium thrust cone which provides mounts for the gimbal block and gimbal actuators. The thrust cone is then connected directly to the hydrogen tank by two bays of 12 member trusses which form a 45-degree cone and tangentially intersect the hydrogen tank's elliptical bulkhead. This 45-degree cone is continued from the hydrogen tank to the structural/meteoroid protective shell with six pairs of struts arranged in truss segmented vees. Lateral support is provided at the forward end of the hydrogen tank by four pairs of struts attached from the structural shell to the elliptical bulkhead and oriented horizontally. The oxygen tank is supported by two truncated cone truss bays. One 12-member truss cone is located tangent to the inboard surface of the torus and slopes inward to its termination point on the thrust cone. The other structural shell. The oxidizer tank is thus supported by two "redundant" load paths. This provides a saddle for the propellant tank which supplies adequate axial, lateral, rotational, and torsional restraint.

All tank support struts and thrust cone truss members are low conductive tubes manufactured of unidirectional fiberglass epoxy. These provide excellent thermal isolation of the cryogenic propellant tanks from the structural shell and engine. Prototype struts of this type have been built and tested by Convair and have shown that design stresses in excess of 100 ksi are acceptable.

Table 4-8 shows a weight breakdown of the 39,000 lb propellant version shown in Figure 4-23.

Table 4-8. LO_2/LH_2 Propulsive Stage - Weight Breakdown ($W_p = 39,000$: Synchronous Equatorial Mission)

Basic Structure		1,342
Cylindrical shell - Al corrugation	976	
Aft conical section - H/C	152	
Aft platform	53	
Thrust section	111	
Miscellaneous	50	
Secondary Structure		108
Equipment supports	60	
Tank supports - $LH_2 = 16$, $LO_2 = 20$	36	
Support fittings	12	
Insulation (1 in. Superinsulation)		234
Shroud	194	
Hangers	20	
Forward closure	12	
Aft closure	8	
Main Propulsion		300
RL10A-3-3A (1)	290	
Mounts/harness	6	
Purge system	4	
Fuel System		536
LH_2 tank	461	
Lines/plumbing	22	
Baffle	20	
Anti-Vortex web	10	
Fill and drain	15	
Heat exchanger plumbing	8	
Oxidizer System		313
LO_2 tank/weldments - torus	234	
Anti-Vortex web (2)	8	
Lines/plumbing	44	
Fill and drain	15	
Shear Web and Baffle (2)	12	
Propellant Loading		10
Propellant Utilization		69
Sensors/harness	30	
Computer	25	
Actuators/plumbing	14	

Table 4-8. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Thrust Vector Control (Electric)			53
Battery		15	
Actuator/supports		33	
Harness		5	
Attitude Control System (N_2O_4/A-50)			110
Tanks (2) (280 lb total cap.)		70	
Mounts/yokes		14	
Plumbing, valves regulators		20	
Harness		6	
Reaction Motors			72
Thrusters (4 clusters)		40	
Mounts/supports		32	
Pressurization System			168
Helium System		127	
(2) Cryogenic bottles/supports	79		
(1) Ambient bottle/supports	20		
Plumbing	28		
Oxidizer System		18	
"O" G vent unit	8		
Ground vent tube	3		
Plumbing/harness	7		
Fuel System		23	
"O" G vent unit	8		
Ground vent	5		
Plumbing/harness	10		
Guidance and Control			183
Guidance		113	
Controls		70	
Electrical System			430
Main System - (4) Fuel Cells		400	
Harness		30	
Telemetry and Instrumentation			104
(1) TLM Unit - Transmitter		50	
(1) P&W Inst. Box		14	
Antennas/landlines		10	
(1) Receiver		30	
Separation System			65
Space Shuttle - truss adapter/docking ring		55	
Payload separation		10	

Table 4-8. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Special Installation - Payload Relays/harness			25
DRY WEIGHT			(4,122)
Residuals			585
Oxidizer - Trapped		72	
Tank	32		
Lines	40		
Fuel - Trapped		22	
Tank	20		
Lines	2		
Propellant Utilization Bias		25	
Gaseous Propellant		398	
GH_2	178		
GO_2	220		
Helium		38	
H_2 tank	13		
O_2 tank	13		
Bottles (3)	9		
RCS	4		
Reaction Control (N_2O_4/A-50)		29	
JETTISON WEIGHT			(4,707)

Propellant Weight Summary

	H_2	O_2	Total
Expendables - Main	6,500	32,500	39,000
Ground Run / Boost Phase — Losses assumed negligible			
Orbit Phase			
Chilldown			
1st burn	-15	-45	-60
2nd burn	-15	-45	-60
3rd burn	-15	-45	-60
4th burn	-15	-45	-60
Engine Start			
1st burn	-5	-27	-32
2nd burn	-5	-27	-32
3rd burn	-5	-27	-32
4th burn	-5	-27	-32

Table 4-8. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Engine Shutdown			
1st burn	-3	-7	-10
2nd burn	-3	-7	-10
3rd burn	-3	-7	-10
4th burn	-3	-7	-10
Coast (7 days)	-168	0	-168
Leakage (7 days)	-1	-3	-4
Main Impulse Expendables	6,239	32,181	38,420
Auxiliary Propellants (N_2O_4/A-50)			
Start Settling (4) 35/30/17/17			99
Shutdown Damping (4)			0
Roll Control during Main Burn (4)			20
Attitude Control - Coast Orientation			24
Initial		4	
5-1/4 hour coast-up		4	
Sync. Eq. Coast - 1		4	
Sync. Eq. Coast - 2		4	
5-1/4 hour coast-back		4	
Rendezvous orbit		4	
Rendezvous (150 ft/sec)			100
Payload Separation			8
Minimum N_2O_4/A-50 Expended			251 lb
Reserve and Contingency*			25
Residual*			4
Minimum N_2O_4/A-50 Required			280 lb
Gross Weight Summary (w/o payload)			
Stage jettison weight			4,707
Expendable propellants - main			39,000
auxiliary			251
helium			17
Gross Weight at Space Shuttle Launch			43,975

*Included in Jettison Weight

4.4.2 STAGE CONFIGURATION - ETR LAUNCH. This configuration is shown in Figure 4-24. Preliminary sizing studies indicate that the stage size required to lift a minimum of 5,000 lb of payload from the inclined 100 n. mi. orbit to a synchronous equatorial orbit and return is approximately 65,000 lb (54,170 lb LO_2 and 10,830 lb LH_2). Launch of this stage from WTR can only be achieved by off-loading propulsive stage propellants.

4.4.2.1 Propellant Tanks. The only LH_2/LO_2 tankage configuration which results in a stage length short enough so that both the stage and payload will fit in the 60 ft space shuttle cargo bay is a stretch version of the 39,000 lb propulsive stage. Stage lengths not utilizing a toroidal oxygen tank configuration are all several feet too long.

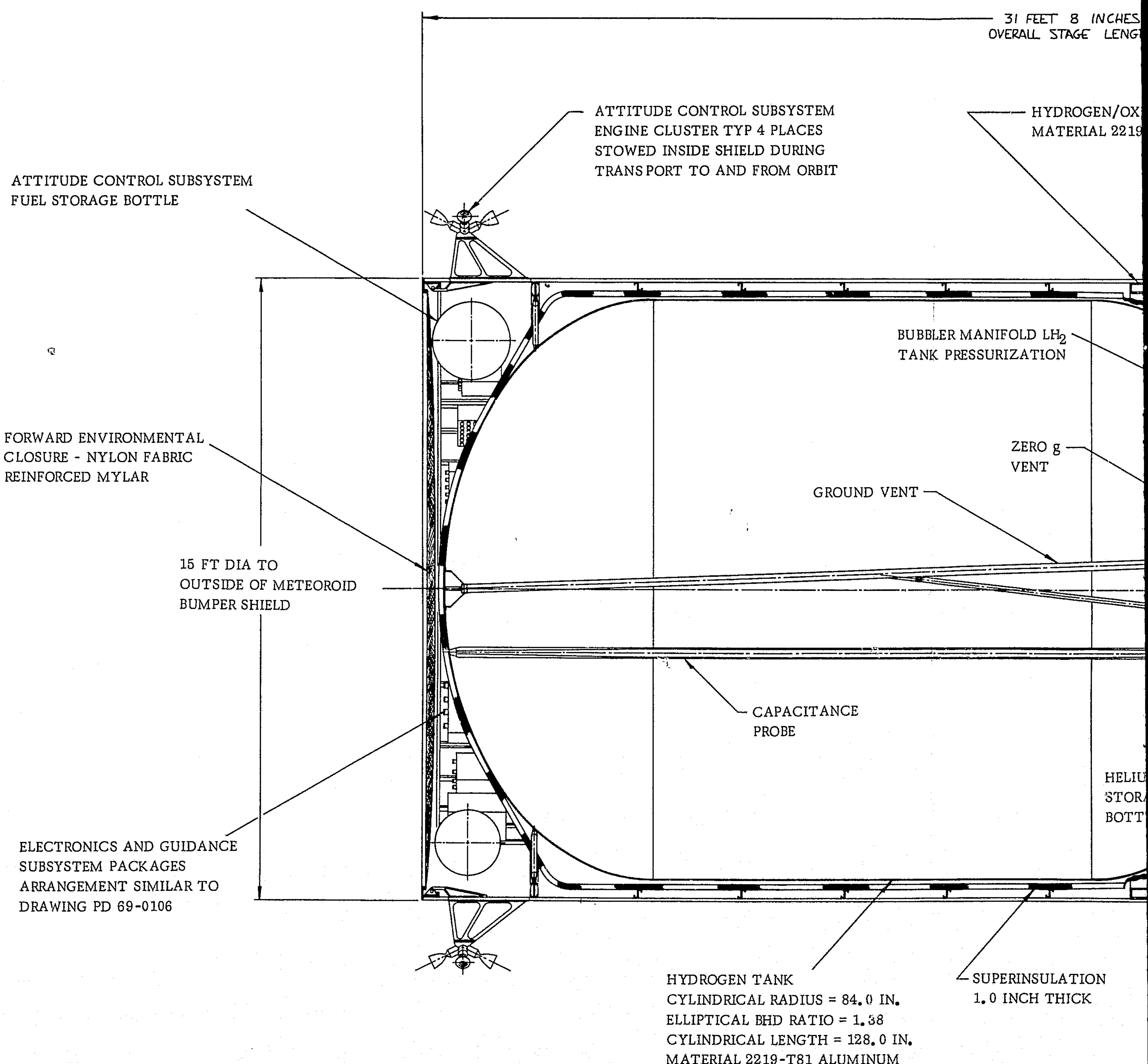
The hydrogen LH_2 tank is identical in shape to the 39,000 lb propulsive stage LH_2 tank, except for its increased cylindrical length. The tank is 14 ft in diameter, with a 128-in. long cylindrical section and 1.38 elliptical bulkheads. The total tank volume is 2,680 ft^3 and it holds 10,830 lb of liquid hydrogen. The tank wall thicknesses were again determined for 2219-T81 aluminum, a 1-g liquid head, and an operating tank pressure of 27 psia. The resulting elliptical bulkhead crown thickness is 0.0365 in. and the cylindrical section is 0.053 in. thick.

The toroidal LO_2 tank chosen for the oxygen propellant is a unique shape which provides the maximum possible volume obtainable for an internally pressurized torus without inducing compressive stresses. It is best described as a zero-hoop-stress meridional torus. If this tank is assumed pressurized by a weightless gas, the hoop membrane forces are zero in all portions of the shell except for a short (38.9 in.) cylindrical section on the outer radius. The resulting toroidal configuration is one which may eliminate the possibility of buckling failures, since the remaining meridional forces will be tensile. The cross-section of revolution has a kidney bean appearance. It has a total length of 84 in., an outside diameter of 14 ft, and an inside diameter of 59.4 in. The torus is also bi-canted to reduce the propellant residuals.

The total tank volume is 814 ft^3, which contains 54,170 lb of liquid oxygen.

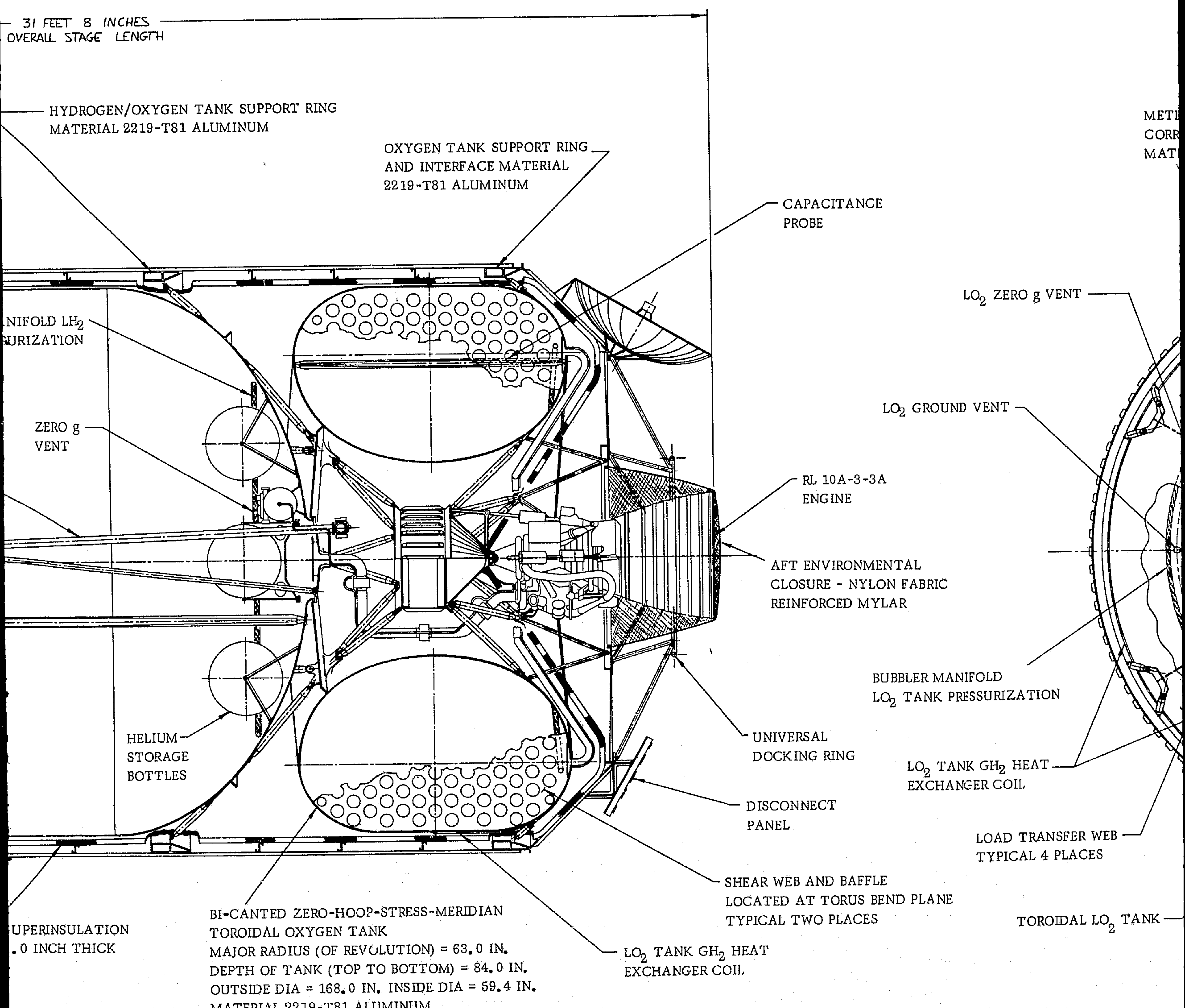
The tank skin gage was determined as before. The thickness on the inside surface of the torus is 0.0512 in. and the cylindrical outer portion is 0.0414-in. thick.

4.4.2.2 Structural Supports. The propellant tank structural support system is similar to the 39,000-lb propulsive stage with several minor exceptions. The first is the configuration of the thrust cone/barrel. The oxygen tank is of sufficient depth that two 45-degree truss cones sloping inboard from tangency points on the tank meet approximately on the vehicle axis. To obtain a distributed load transfer in this region, a 30-in. diameter barrel is inserted between the two truss cones. The thrust cone is identical to the 39K model and is attached to the aft end of this barrel. The second difference is the configuration of the truss cone which is tangent to the aft bulkhead of the hydrogen tank. This structure of six strut pairs is arranged in three bays on a 45-degree cone,



FOLDOUT FRAME 1

Figure 4-24. Propulsive Stage, 65,000 lb Propellant



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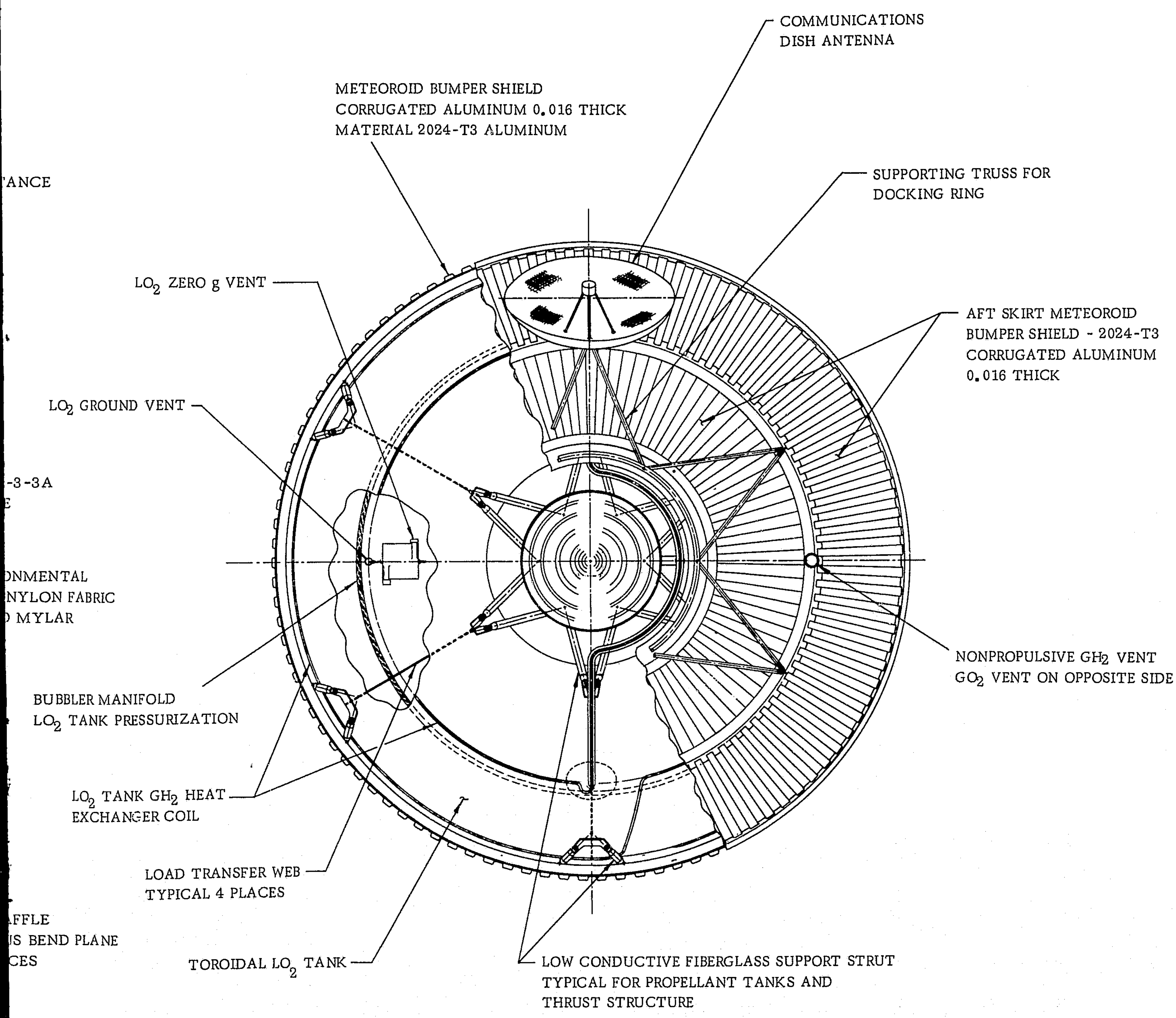

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as was the similar structure on the 39,000-lb vehicle. This cone, however, passes tangent to and attaches to the oxidizer tank. This is done to provide adequate lateral support for the longer tank and to decrease the stage length by moving the oxygen tank as far forward as possible. One additional difficulty results from transmitting this load path through the oxygen tank. Because the tank is bi-canted, this truss does not lie exactly within a 45-degree conical surface, thereby inducing some slight bending loads into both propellant tanks at the tangency point. A ring, integral with the propellant tank, has been added on both tanks to properly react these loads.

4.4.2.3 Structural Shell. The structural shell for the 65,000-lb propulsive stage is again similar to the 39,000-lb shell. The structure is primarily designed to meet meteoroid penetration and structural load requirements. Here again, a shielding factor of 0.5 was used. For the appropriate tank wall gages and meteoroid bumper shields above 0.013, the probability of no penetration of the tanks for a seven-day space residence is 0.9958.

The shell is corrugated 2024-T3 aluminum 0.016-in. thick. A 0.025-in. doubler is used under this bumper shield aft of the thrust cone ring to react tank support ring loads. The frames and their spacing are again dictated by general stability criteria and handling considerations. They are 3.0-in. deep Z sections 0.040-in. thick, spaced 30 in. apart. The forward end of the hydrogen tank is again supported by four strut pairs which attach to an I cross-section frame on the structural shell.

The propellant tanks and thrust cone are supported by struts which attach to the shell at two rings. These rings are each three in. wide and 12 in. deep, with 0.125-in. thick walls. The aft ring also serves as the interface with the orbiter support structure.

The cargo bay interface support beam and longerons shown on the 39,000-lb propulsive stage are not included in the 65,000-lb design. The configuration shown employs a distributed load transfer to an interface support structure in the orbiter, rather than a point load transfer from the propulsive stage. However, if a non-optimum load distribution occurs in the orbiter interface support structure, some small longerons may be required between the tank support rings.

4.4.2.4 Thermal Protection System. The superinsulation and purge system used on the 65,000 lb propulsive stage is an identical scaled-up version of the 39,000 lb configuration. A 1-in. multi-layer radiation shield blanket encapsulates both propellant tanks. The vehicle structural shell serves as the purge control volume.

The weights summary for this stage is given in Table 4-9.

4.4.3 SPACE SHUTTLE/PROPULSIVE STAGE INTERFACES. One of the major areas where the propulsive stage differs from standard high energy upper stage vehicles is the interface definition between the propulsive stage and orbiter. This subject can be divided into two categories. The first deals with the fluid interfaces and the second

Table 4-9. LO_2/LH_2 Propulsive Stage - Weight Breakdown
($W_p = 65,000$ lb: Synchronous Equatorial Mission)

Basic Structure		1,676
Cylindrical shell	1,331	
Aft conical section	82	
Aft skirt - platform	51	
Thrust section	144	
Disconnect panel	15	
Miscellaneous	53	
Secondary Structure		140
Equipment supports	60	
Tank supports - LH_2 = 27, LO_2 = 33	60	
Support fittings	20	
Insulation (1" Superinsulation)		354
Shroud	302	
Hangers	30	
Forward closure	12	
Aft closure	8	
Main Propulsion		300
RL10A-3-3A (1)	290	
Mounts/harness	6	
Purge system	4	
Fuel System		827
LH_2 tank	746	
Lines/plumbing	24	
Baffle	20	
Anti-Vortex web	10	
Fill & drain	17	
Heat exchanger plumbing	10	
Oxidizer System		583
LO_2 tank/weldments - torus	476	
Anti-Vortex web (2)	8	
Lines/plumbing	51	
Fill & drain	15	
Sheer webs & baffles (2)	15	
Load Transfer webs	18	

Table 4-9. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Propellant Loading			17
Propellant Utilization			90
Sensors/harness		50	
Computer		25	
Actuators/plumbing		15	
Thrust Vector Control (Electric)			53
Battery		15	
Actuator/supports		33	
Harness		5	
Attitude Control System (N_2O_4/A-50)			154
Tanks (2) (400 lbs total cap.)		100	
Mounts/yokes		20	
Plumbing, valves regulators		26	
Harness		8	
Reaction Motors			72
Thrusters (4 clusters)		40	
Mounts/supports		32	
Pressurization System			260
Helium System		183	
(3) Cryogenic bottles/supports	120		
(1) Ambient bottle/supports	25		
Plumbing	38		
Oxidizer System		32	
"O" G vent unit	8		
Ground vent tube	6		
Plumbing/harness	10		
Bubbler manifold	8		
Fuel System		45	
"O" G vent unit	8		
Ground vent	10		
Plumbing/harness	15		
Bubbler manifold	12		
Guidance & Control			183
Guidance		113	
Controls		70	
Electrical System			430
Main system - (4) Fuel cells		400	
Harness		30	

Table 4-9. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Telemetry & Instrumentation			109
(1) Transmitter		50	
(1) P&W inst. box		14	
(1) Receiver		30	
Antenna & support		15	
Separation System			95
Orbiter - truss adapter/docking ring		85	
Payload separation		10	
Special Installation - Payload			25
Relays/harness			
DRY WEIGHT			(5,368)
Residuals			905
Oxidizer - Trapped		77	
Tank	32		
Lines	45		
Fuel - Trapped		24	
Tank	20		
Lines	4		
P. U. Bias		40	
Gaseous Propellant		664	
GH_2	297		
GO_2	367		
Helium		59	
H_2 tank	21		
O_2 tank	20		
Bottles (4)	13		
RCS	5		
Reaction Control (N_2O_4/A-50)		41	
JETTISON WEIGHT			(6,273)

Propellant Weight Summary

	H_2	O_2	Total
Expendables - Main	10,833	54,167	65,000
Ground Run / Boost Phase } Losses assumed negligible			

Table 4-9. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

	H_2	O_2	Total
Orbit Phase			
Chilldown			
1st burn	-15	-45	-60
2nd burn	-15	-45	-60
3rd burn	-15	-45	-60
4th burn	-15	-45	-60
Engine Start			
1st burn	-5	-27	-32
2nd burn	-5	-27	-32
3rd burn	-5	-27	-32
4th burn	-5	-27	-32
Engine Shutdown			
1st burn	-3	-7	-10
2nd burn	-3	-7	-10
3rd burn	-3	-7	-10
4th burn	-3	-7	-10
Coast (7 days)	-168	0	-168
Leakage (7 days)	-1	-3	-4
Main Impulse Expendables	10,572	53,848	64,420

Auxiliary Propellants (N_2O_4/A-50)

Start Settling (4) 56/48/27/27		158
Shutdown Damping (4)		0
Roll Control during Main Burn (4)		27
Attitude Control - Coast Orientation		36
Initial	6	
5-1/4 hour coast-up	6	
Sync. Eq. Coast - 1	6	
Sync. Eq. Coast - 2	6	
5-1/4 hour coast-back	6	
Rendezvous orbit	6	
Rendezvous (150 ft/sec)		125
Payload Separation		13
Minimum N_2O_4/A-50 Expended		359

Table 4-9. LO_2/LH_2 Propulsive Stage - Weight Breakdown, Contd

Reserve & Contingency*	**35**
Residual*	**6**
Minimum N_2O_4/A-50 Required	**400 lb**
Gross Weight Summary (w/o payload)	
Stage jettison weight	**6,273**
Expendable propellants - main	**65,000**
- auxiliary	**359**
Helium - expendable	**22**
Gross Weight at Booster Launch	**71,654**

***Included in Jettison Weight**

deals with the mechanical/structural interfaces. The interface discussion included herein is essentially independent of propulsive stage size and is applicable to both previously described stages.

4.4.3.1 Fluid System Interfaces. Two basic concepts have been investigated. The first fully integrates the propulsive stage fluid servicing lines with the orbiter, while the second system has completely independent service systems. There are other concepts which are combinations of these two, but for relative comparison purposes these two types of systems have been evaluated.

Separated Systems (Figure 4-25). Two arrangements are shown for the separate servicing line concept. Figure 4-26 locates the internal disconnect panel forward of the transfer stage oxidizer tank. This location permits the mounting of the overboard ground umbilical fittings on the same umbilical panel used by some of the orbiter circuits. A limited space envelope for the support and activation of the internal disconnect panel is the primary disadvantage.

The second method shown in Figure 4-27 locates the internal disconnect panel aft of the transfer stage oxidizer tank. This location provides larger space envelopes for the panel assembly, flex line sections, activation mechanisms, and structural supports. The overboard umbilical connections, however, are mounted on a separate panel which may require additional weight and ground actuation systems. Tradeoffs exist when considering common or separate overboard umbilical panels. In some cases, the vehicles may be penalized with additional line lengths to accomplish common panels. Efforts should be made to place the burdens on the ground systems rather than on the vehicles.

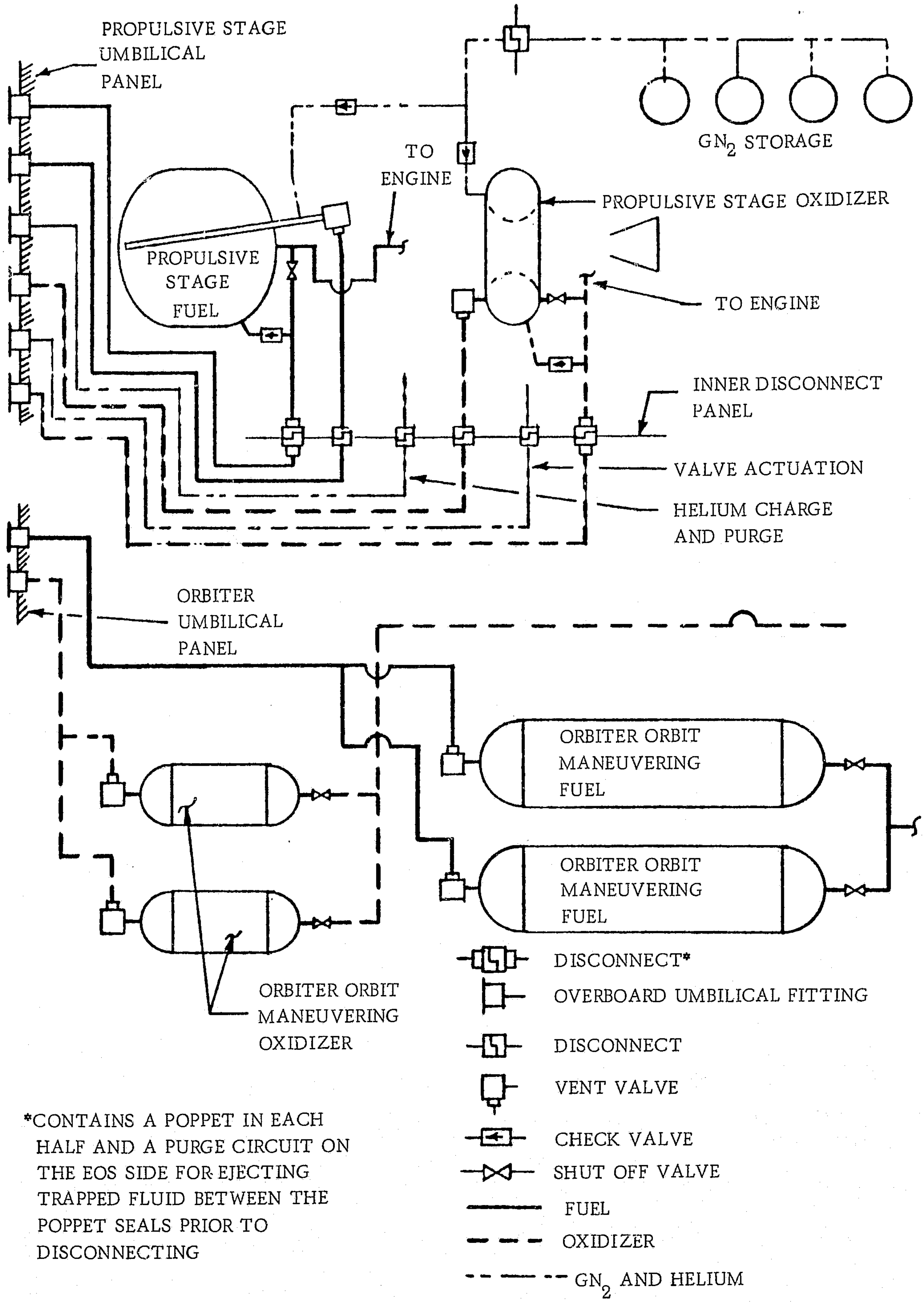

Figure 4-25. Propulsive Stage/Orbiter Interface Schematic, Independent Systems

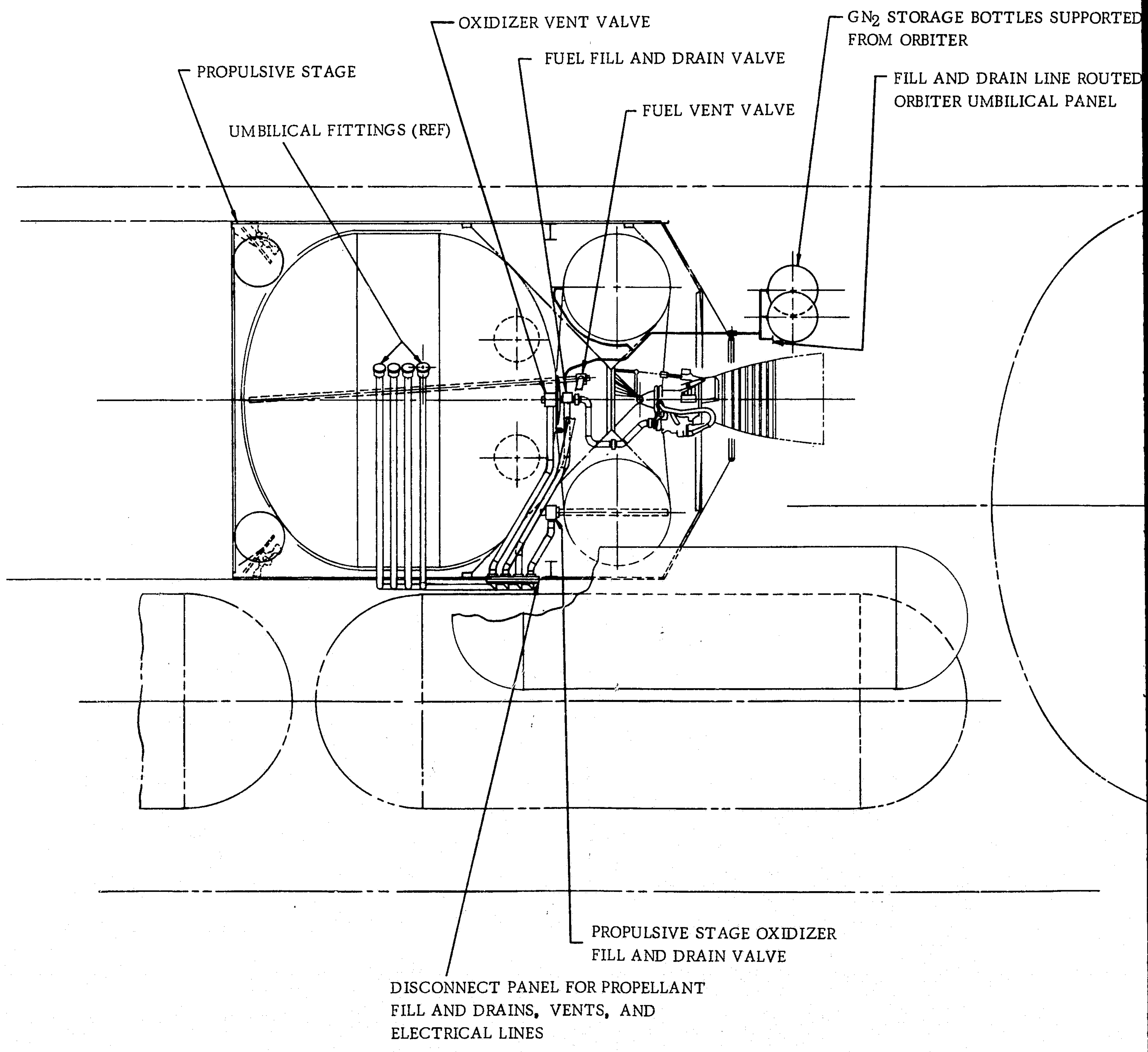

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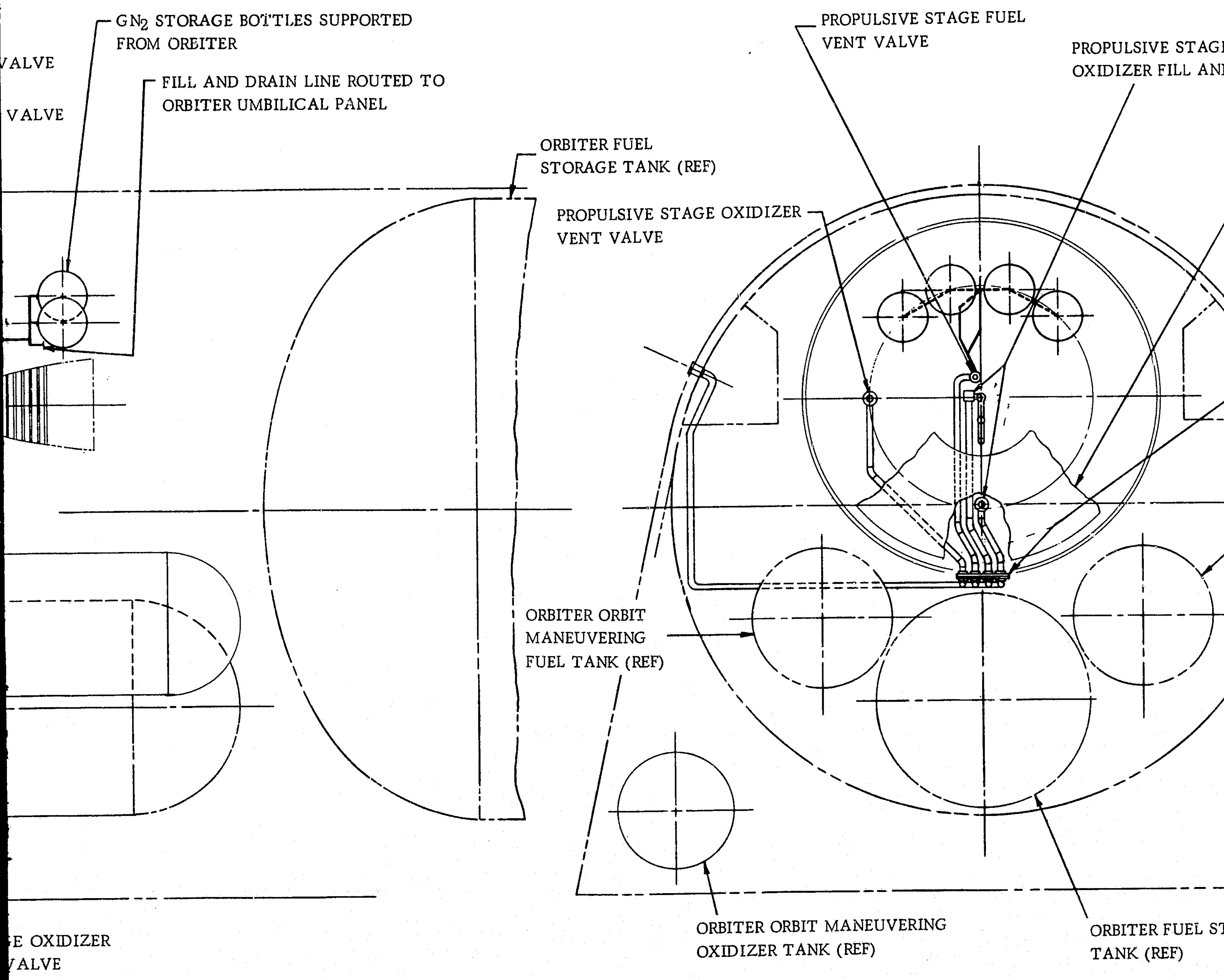
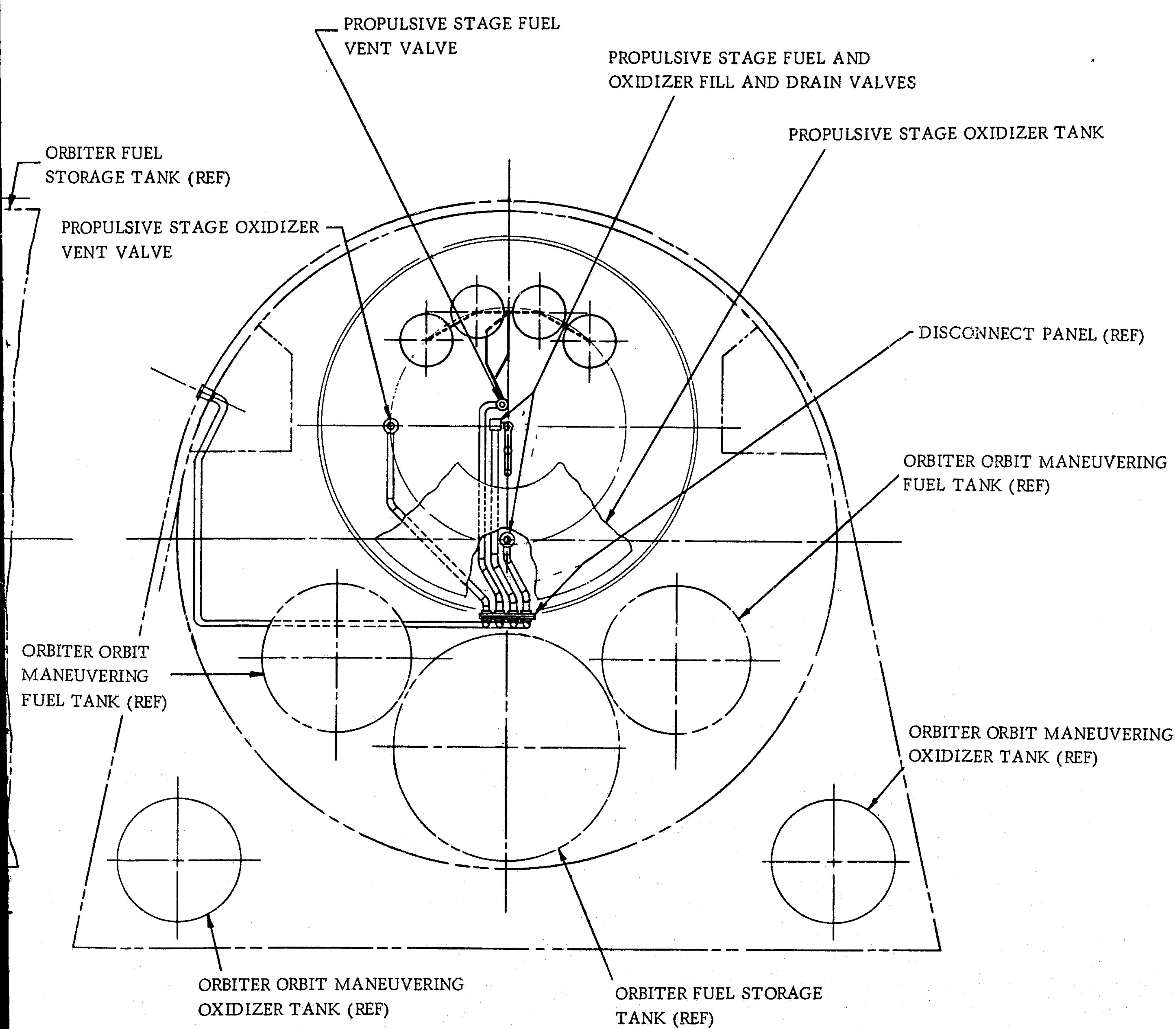
Figure 4-26. Propulsive Stage/Orbiter Interfaces, Independent Propellant System



ropellant System



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FOLDOUT FRAME 3

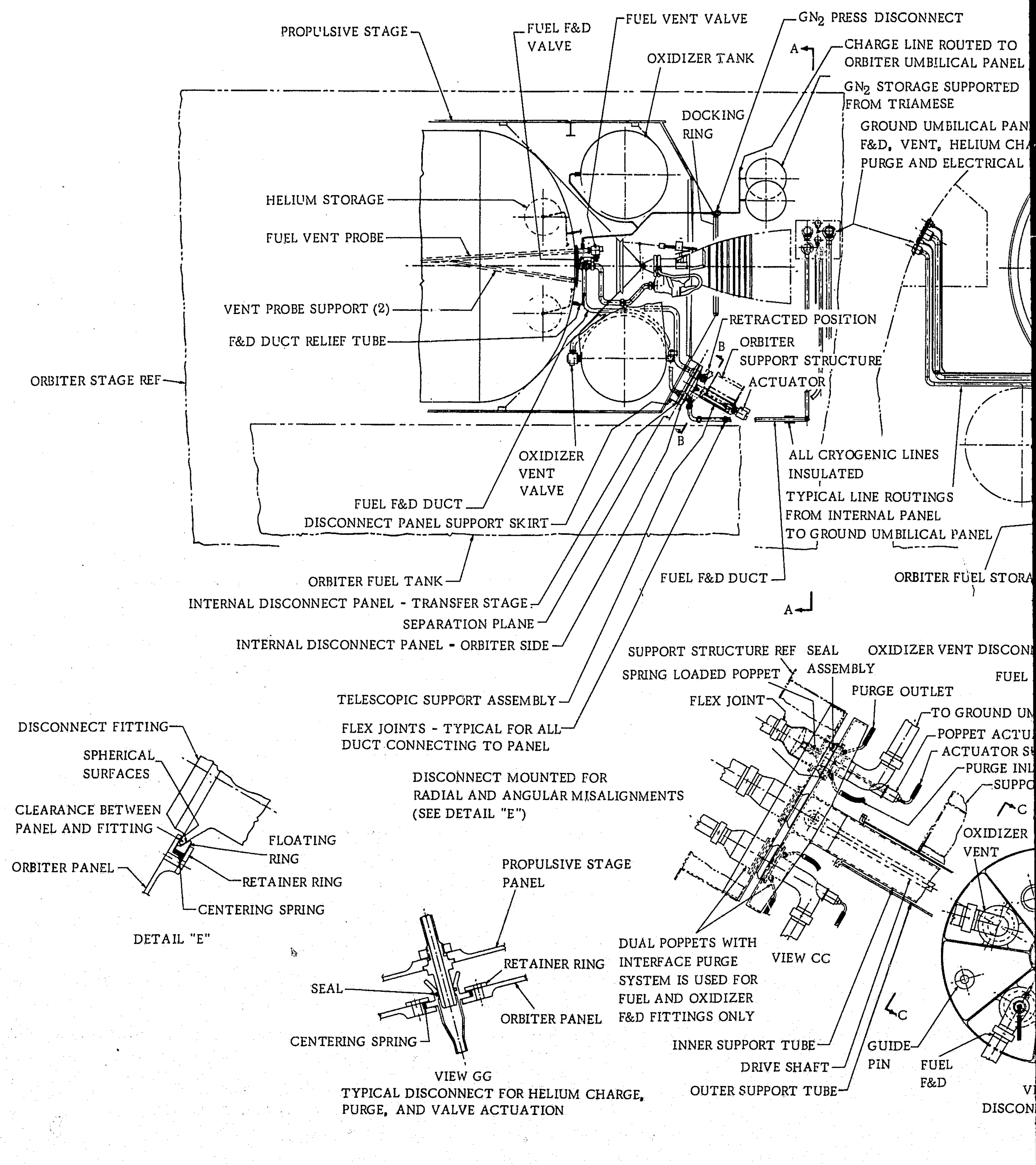

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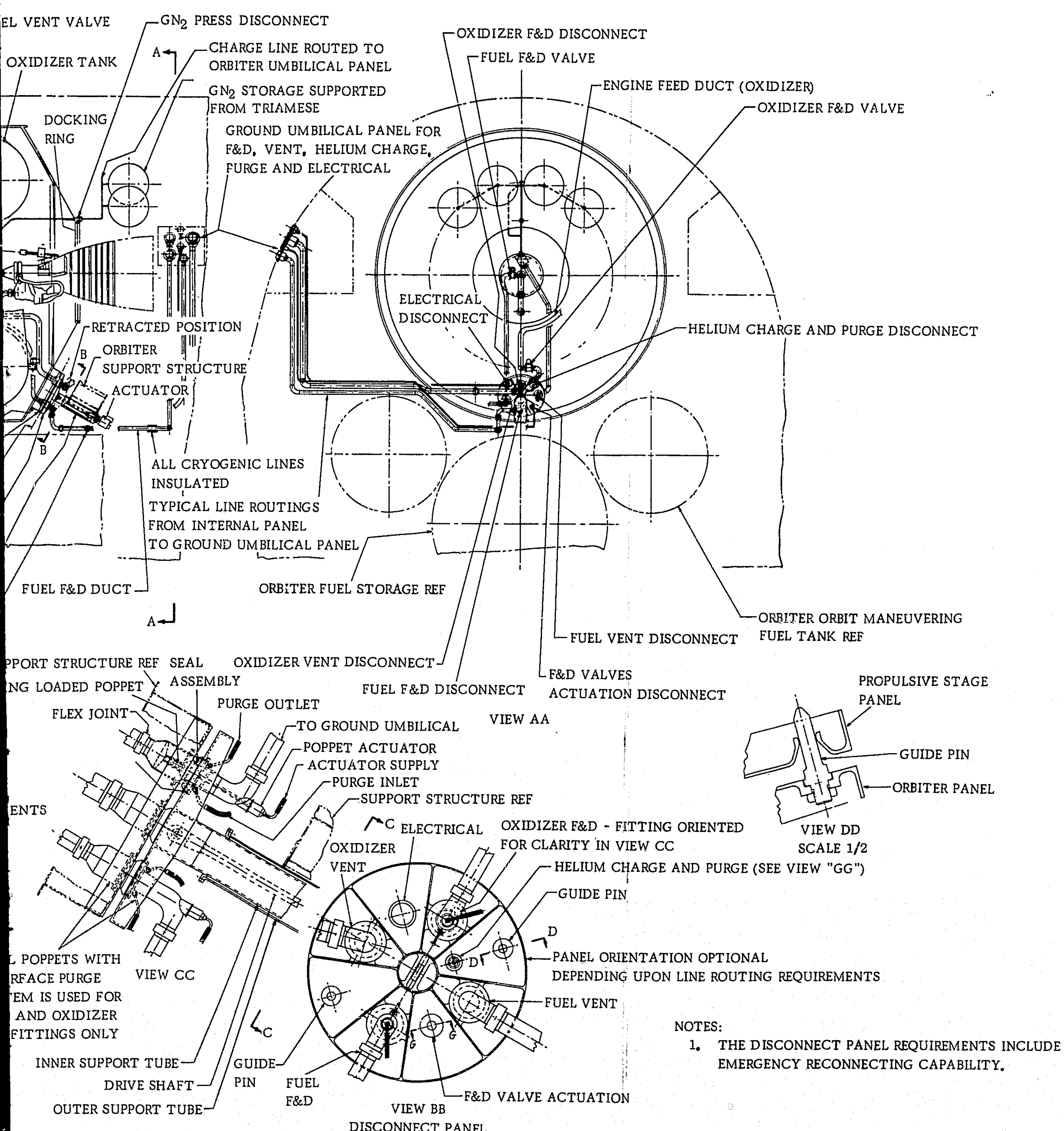
Figure 4-27. Propul



NOTES:

1. THE DISCONNECT PANEL REQUIREMENTS INCLUDE EMERGENCY RECONNECTING CAPABILITY.

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Figure 4-27. Propulsive Stage/Orbiter Interfaces, Independent Propellant Systems

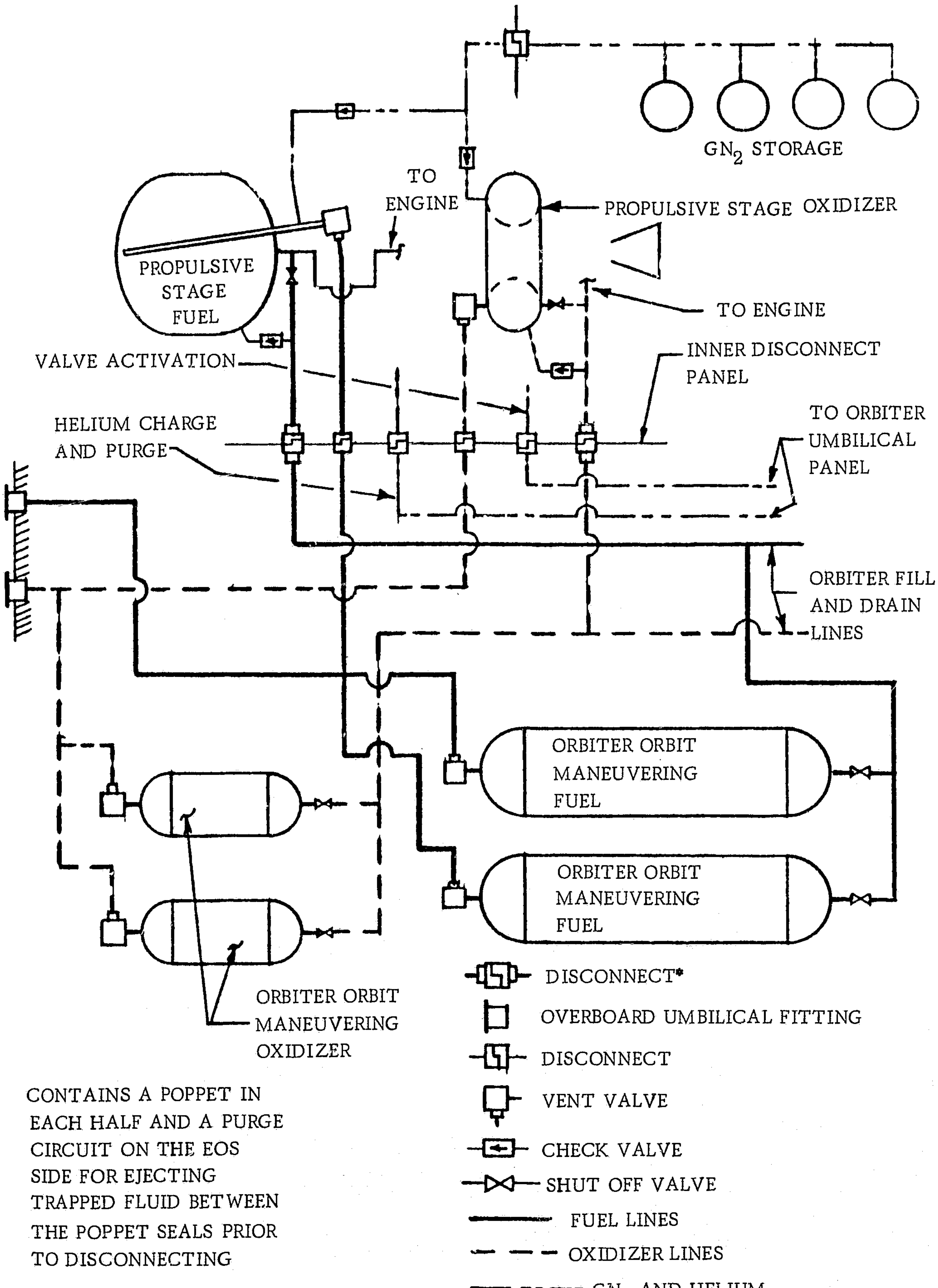
Integrated Systems. In addition to the usual need for optimum routing of lightweight, flexible propellant ducts the design must provide for separation from and reloading into the orbiter, propellant jettisoning in case of an emergency, and ground vent capability.

The fill and drain installation conveys propellants from the ground umbilical, through an internal disconnect panel, through the orbiter stage, to the transfer stage propellant tanks. The transfer stage fill-and-drain systems and vent systems are interconnected with the orbiter circuits (see Figure 4-28). An installation drawing is shown in Figure 4-29. For emergency conditions, the transfer stage propellants may be dumped into the orbiter stage and subsequently burned through the main engines. The vent ducts are routed through the orbiter circuits to the ground umbilical panel. The basic concepts of the separated system disconnect panel, GN_2 tank pressurization, and purge requirements are similar to that of the separate system.

4.4.3.2 Propulsive Stage/Orbiter Structural Interfaces. Structural attachments between the propulsive stage and the orbiter are required which adequately transmit loads between the two vehicles while still allowing for deployment and recapture of the orbital stage.

The structural attachment is complicated by the configuration of the orbiter cargo bay. The only structural members available to carry the propulsive stage loads are two longerons, each running the length of the cargo bay along each side. Thus, the distributed axial load about the 47-ft circumference of the propulsive stage must be transferred to two points in the orbiter. This requires the addition of special structure to transfer the loads which may be included either as a part of the propulsive stage or a part of the orbiter.

A method of adding the special structure to the cargo bay of the orbiter is shown in Figures 4-30 and 4-31. This structure consists of a distributed load ring which mates with the aft adapter ring of the propulsive stage and a set of oblique skin stringer frame cone segments to transfer this distributed load to two axial load pin sockets located approximately six feet aft of the ring. This configuration is chosen for maximum structural efficiency (minimum weight) and is about 75 percent effective in transmitting the two point loads to a fully distributed loading on the adapter ring. Latches are included on the mating ring to engage and hold the propulsive stage during orbiter maneuvering, reentry, and landing. These 24 structural latches (shown in Figure 4-30) must be reusable for multiple deployments and recoveries of the propulsive stage vehicle. The latches shown are normally open and pneumatically actuated to the holding position. This provides a fail-open mode so that the propulsive stage can still be deployed or recovered should one or more latches become inoperable. The use of an external interface support structure of this type is not limited to the transport of the propulsive stage. All projected orbiter payloads will have a load transfer problem identical to the propulsive stage. This structure will work equally well for these other payloads.



*CONTAINS A POPPET IN EACH HALF AND A PURGE CIRCUIT ON THE EOS SIDE FOR EJECTING TRAPPED FLUID BETWEEN THE POPPET SEALS PRIOR TO DISCONNECTING

Figure 4-28. Propulsive Stage/Orbiter Interface Schematic, Integral Systems Orbiter

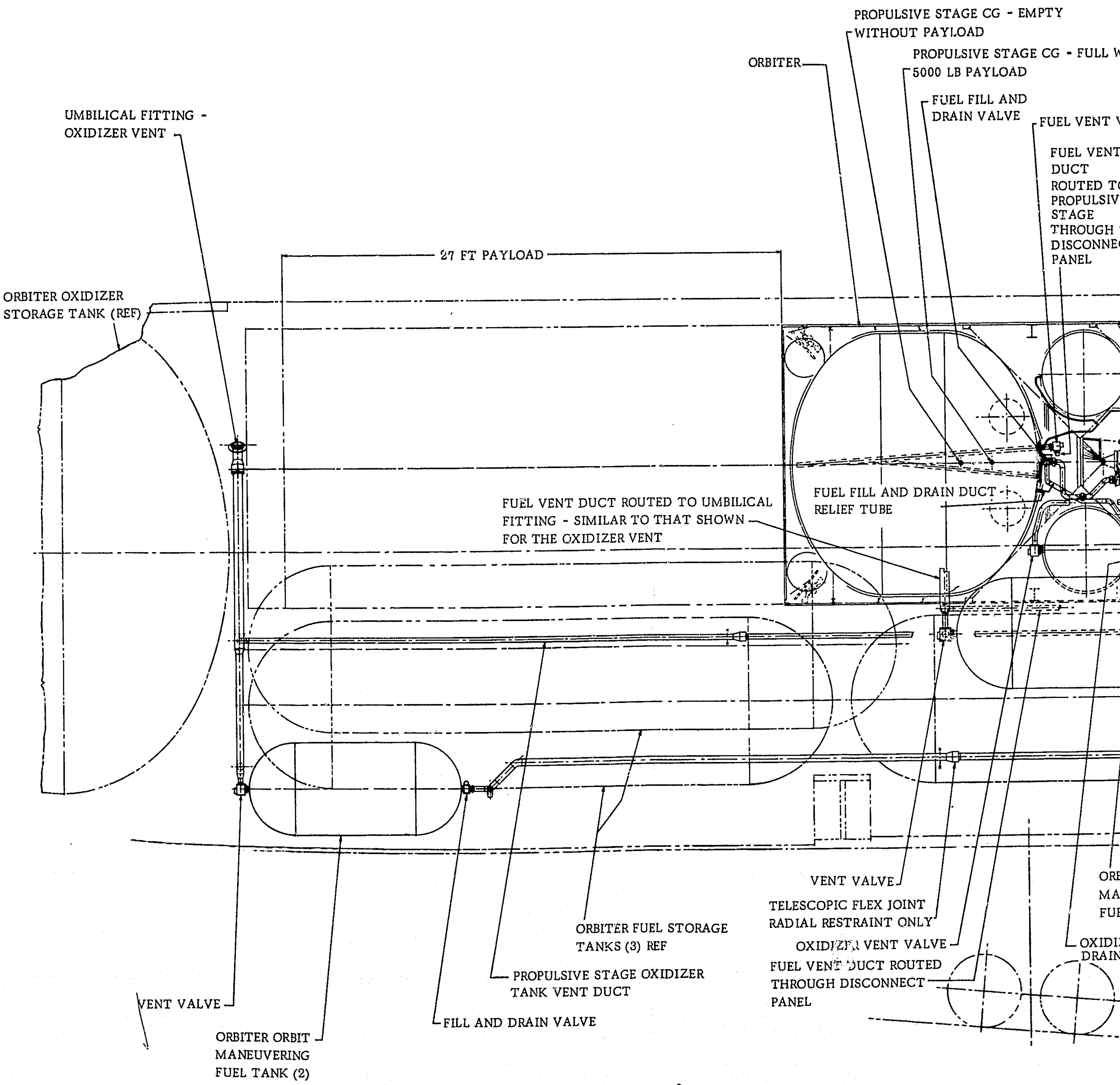


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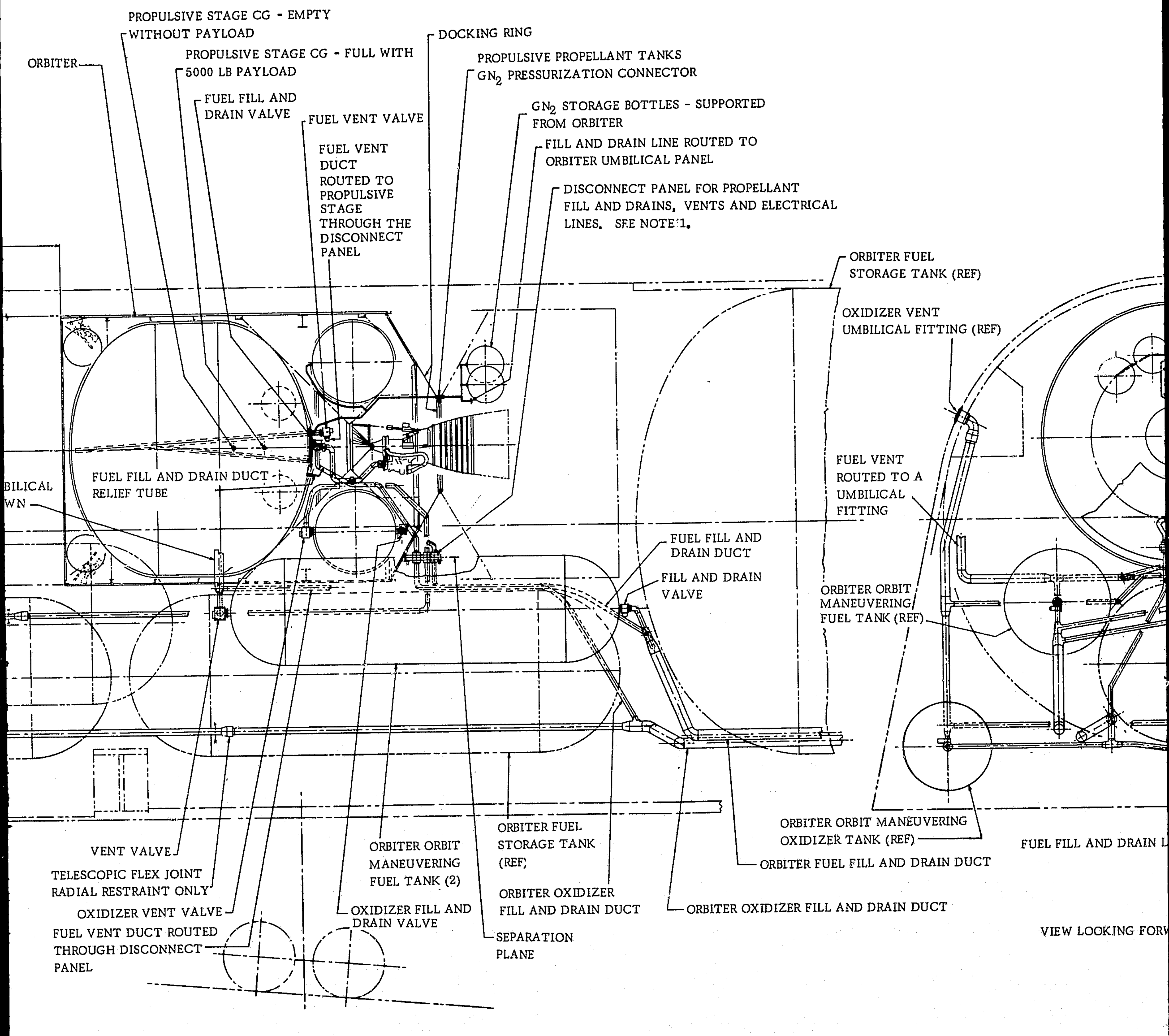
Figure 4-29. Propulsive Stage/Orbiter Interfaces, Integral Propellant Systems



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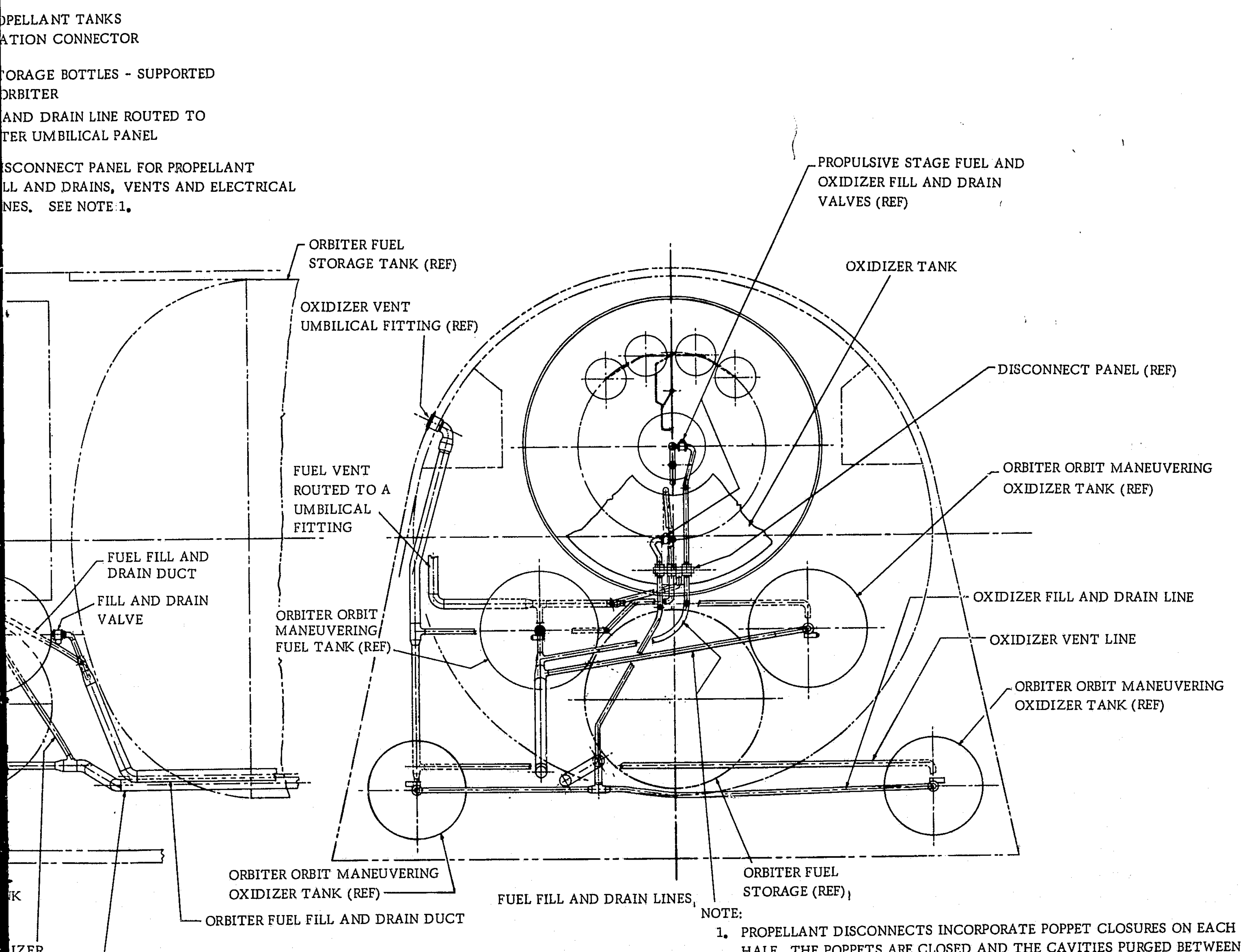
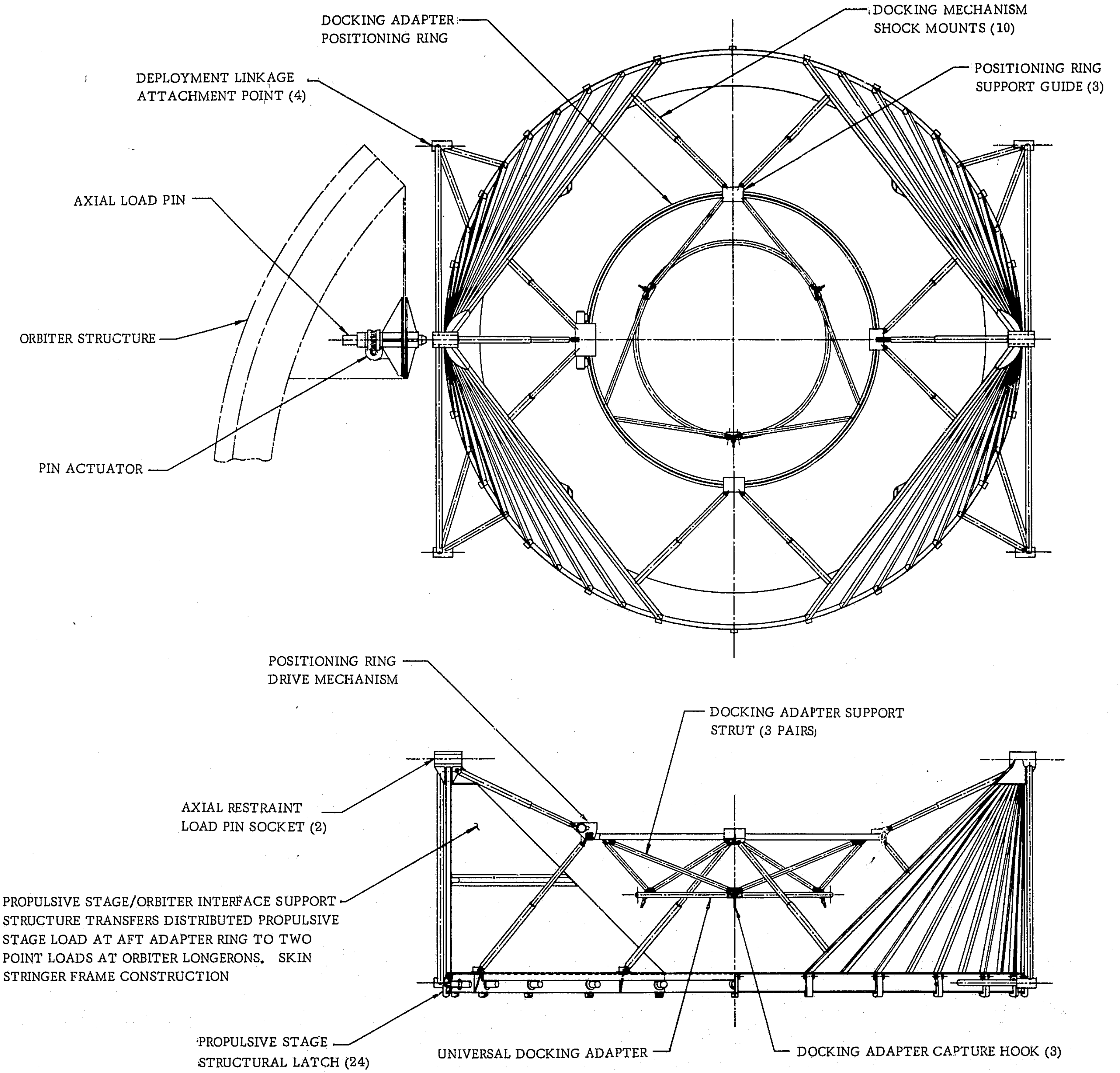


NOTE:

1. PROPELLANT DISCONNECTS INCORPORATE POPPET CLOSURES ON EACH HALF. THE POPPETS ARE CLOSED AND THE CAVITIES PURGED BETWEEN CLOSURES PRIOR TO DISENGAGEMENT. THE SUPPORT PANELS INCLUDE FEATURES FOR SEPARATING THE DISCONNECT FITTINGS PRIOR TO DEPLOYMENT OF THE TRANSFER STAGE.

VIEW LOOKING FORWARD



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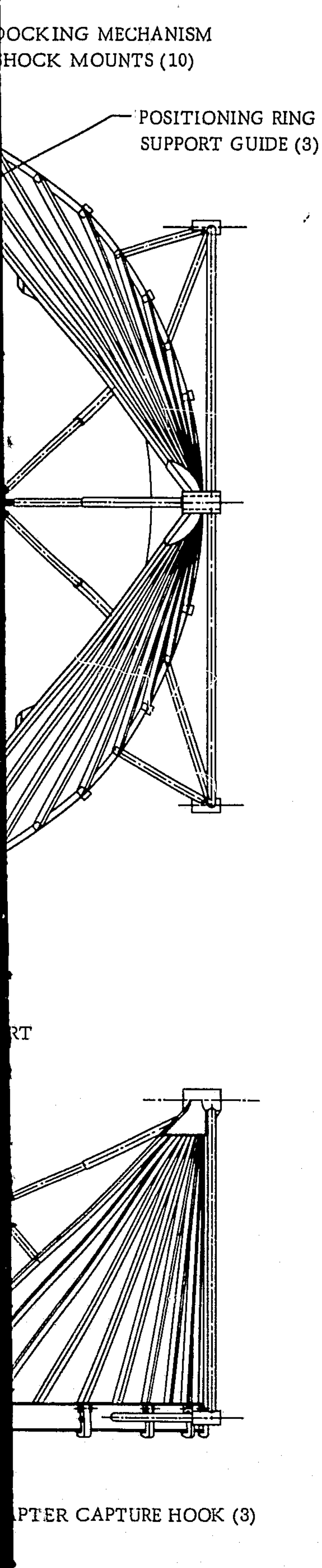

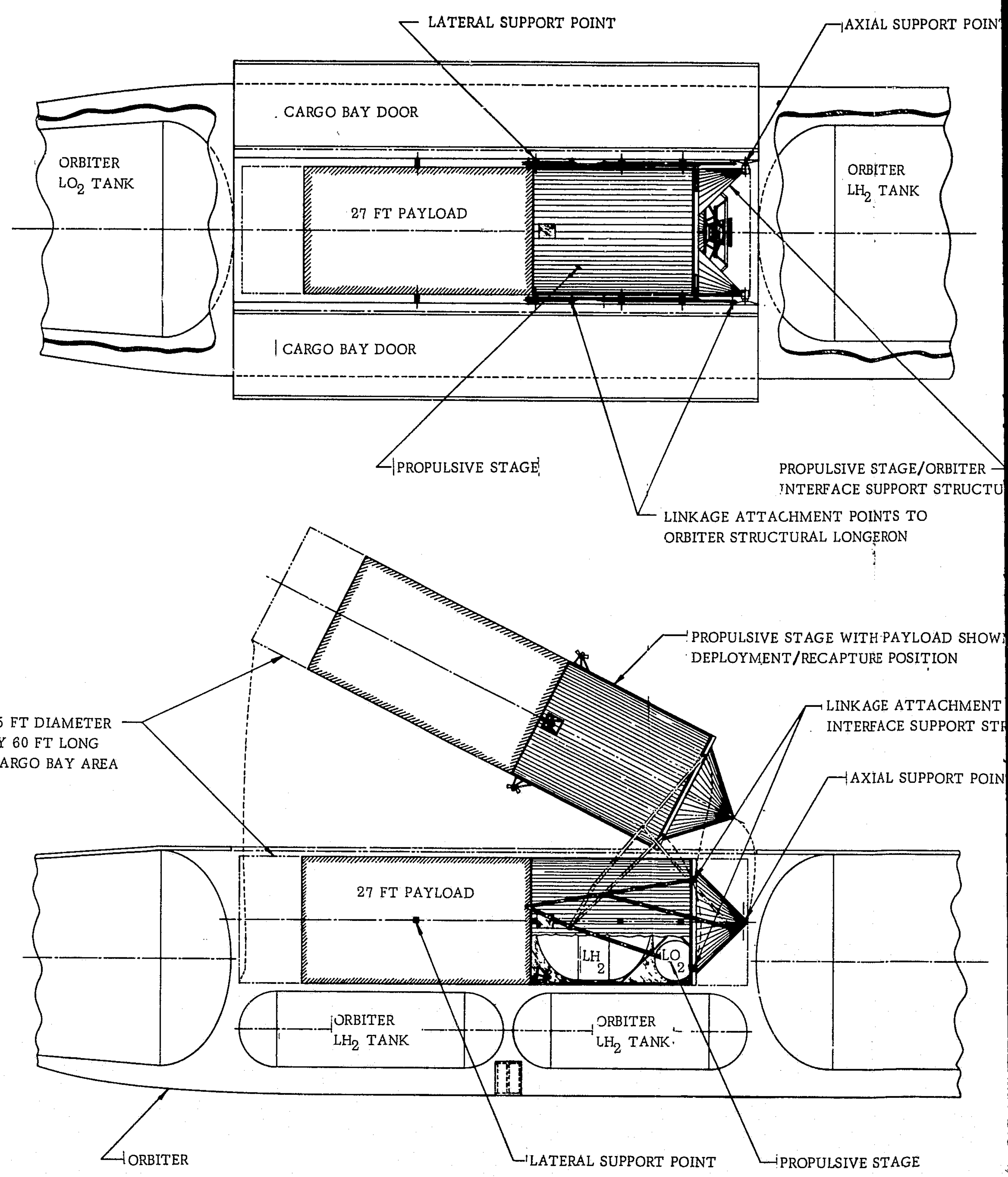

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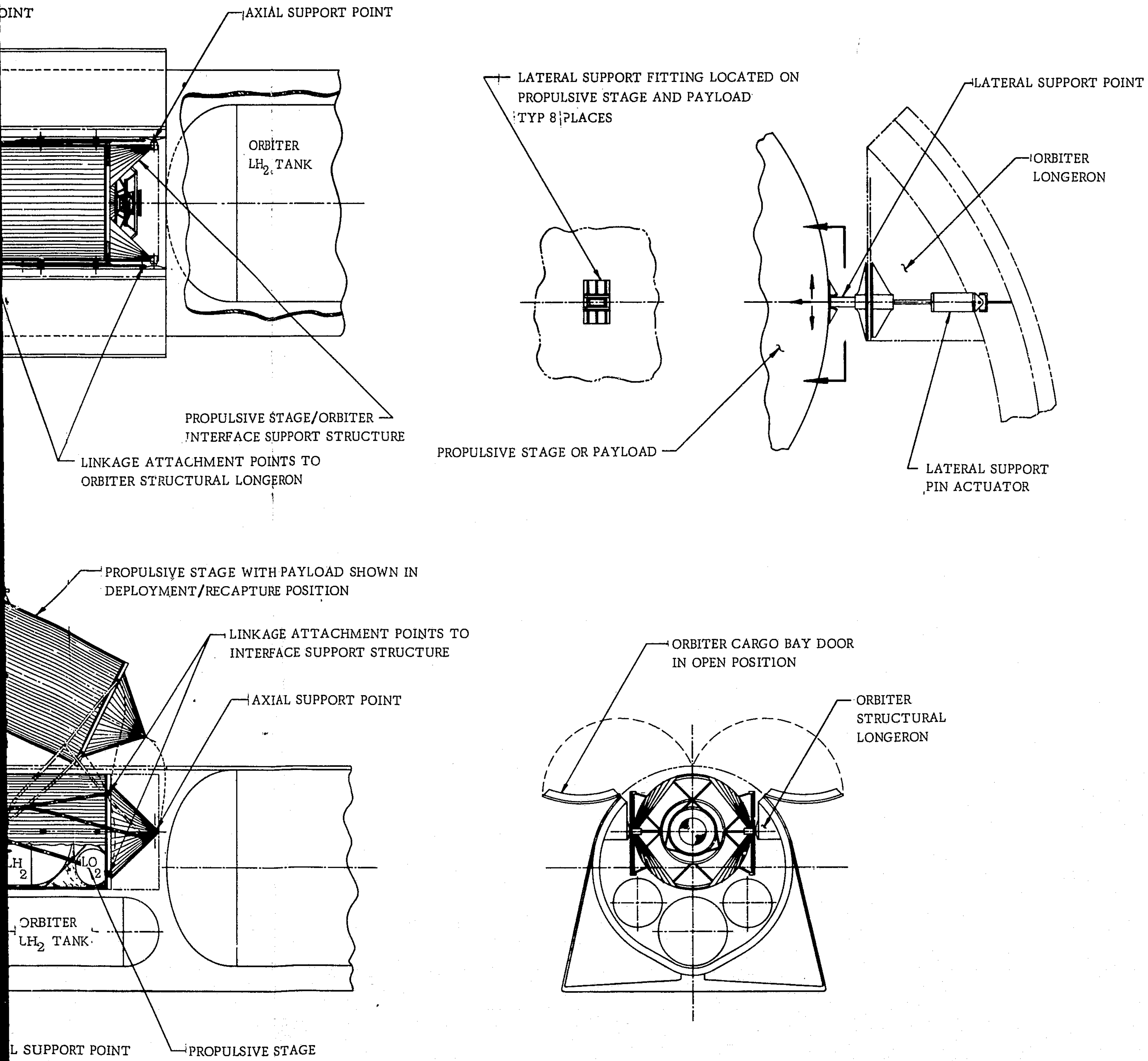

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Figure 4-30. Propulsive Stage/Orbiter Structural Interface, 39,000 lb Propellant Weight

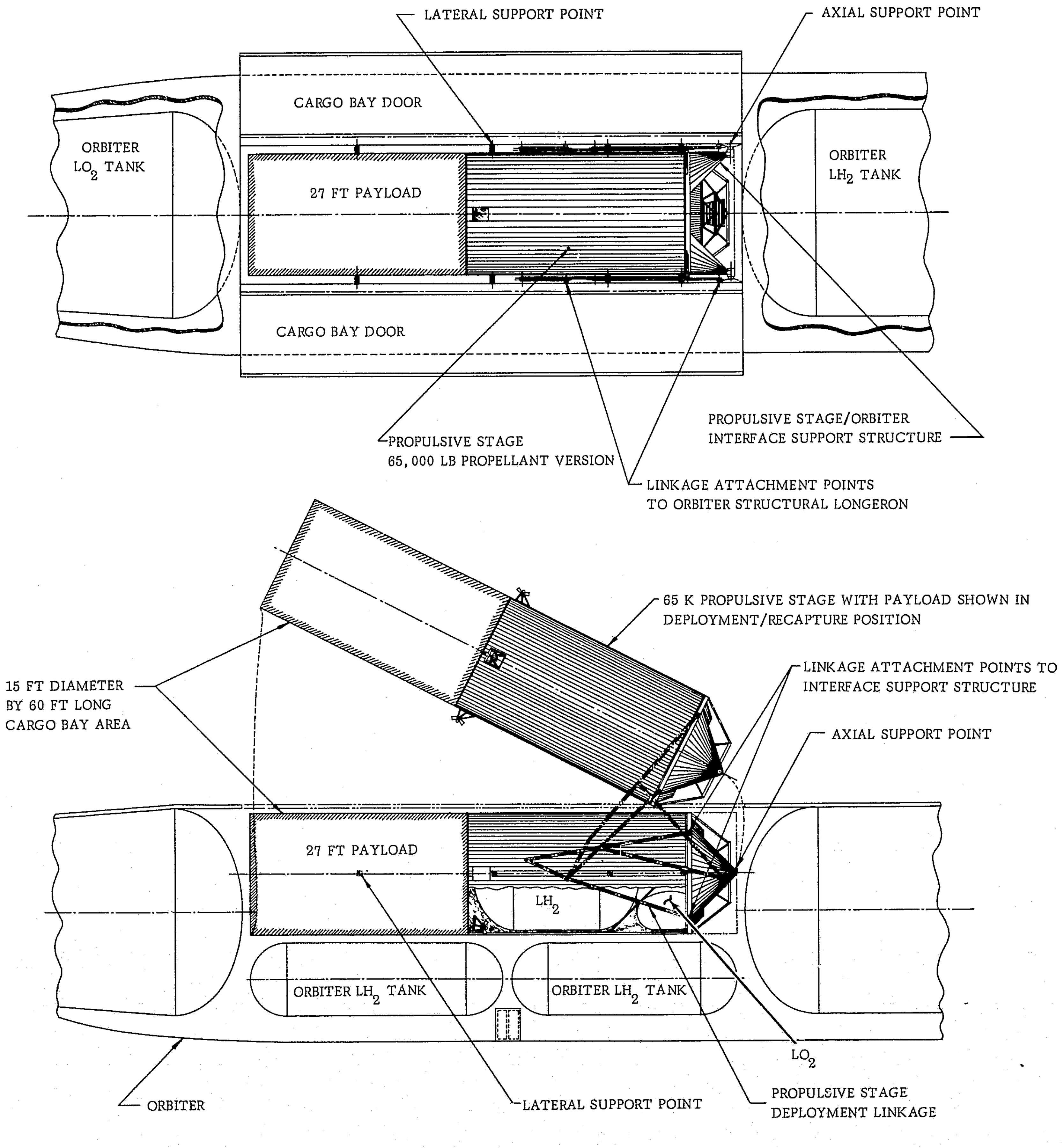


Figure 4-31. Propulsive Stage/Orbiter Structural Interface, 65,000 lb Propellant Weight

The load transfer between the propulsive stage and the orbiter is accomplished by two axial load pins and four pairs of lateral load pins. These 10 load pins are supported from the orbiter longeron structure on each side of the cargo bay and are actuated by either electrical motor-driven ball-screws or pneumatic cylinders. The actuators must retract the aft lateral pins far enough to allow unrestricted movement of the propulsive stage deployment linkage. The axial pins mate with the sockets located on the support structure and are loaded in single shear. The lateral support pins are located one on each side of the two propulsive stage propellant tank support rings, the forward end of the propulsive stage, and the payload center of gravity. They provide resistance to vertical and side loads. These 10 pins provide complete stability for the propulsive stage in the cargo bay during space shuttle launch, orbital maneuvering, reentry, and landing.

4.4.4 PERFORMANCE. The performance of the propulsive stage on the baseline mission and alternate missions is summarized below.

The initial performance analysis for the propulsive stage is based upon varying propellant type and propellant weights for the purpose of rough sizing. An oxygen/hydrogen stage at a gross weight of 50,000 lb is the first configuration selected for the purpose of more detailed performance analysis. The propellants investigated, with their specific impulses, are:

Propellant Type	Specific Impulse (sec)
Fluorine/Hydrogen, F_2/H_2	460
Oxygen/Hydrogen, O_2/H_2	444, 454
Oxygen Difluoride/Methane, OF_2/CH_4	395
Nitrogen Tetroxide/Aerozine 50, N_2O_4/A-50	292, 340

Performance is based upon a flight performance reserve (FPR) at 2 percent of stage total ideal velocity. Stage burnout weight (scaling laws) for the sizing study is presented in Figure 4-32 as a function of stage propellant weight.

Two mission modes and four vehicle operational modes were considered during the study. The two mission modes considered were the synchronous equatorial circular orbit mission and the synchronous inclined circular orbit mission. Vehicle operational modes include payload placement (payload is placed into orbit by the vehicle) with (1) an expendable vehicle (vehicle does not return to the orbiter) and (2) a recoverable vehicle (vehicle returns to the orbiter). The recoverable vehicle is also used for (3) payload retrieval (vehicle without payload goes to orbit to retrieve an orbiting payload), and (4) payload placement and retrieval. Table 4-10 identifies the performance curve for each of the mission and operational modes.

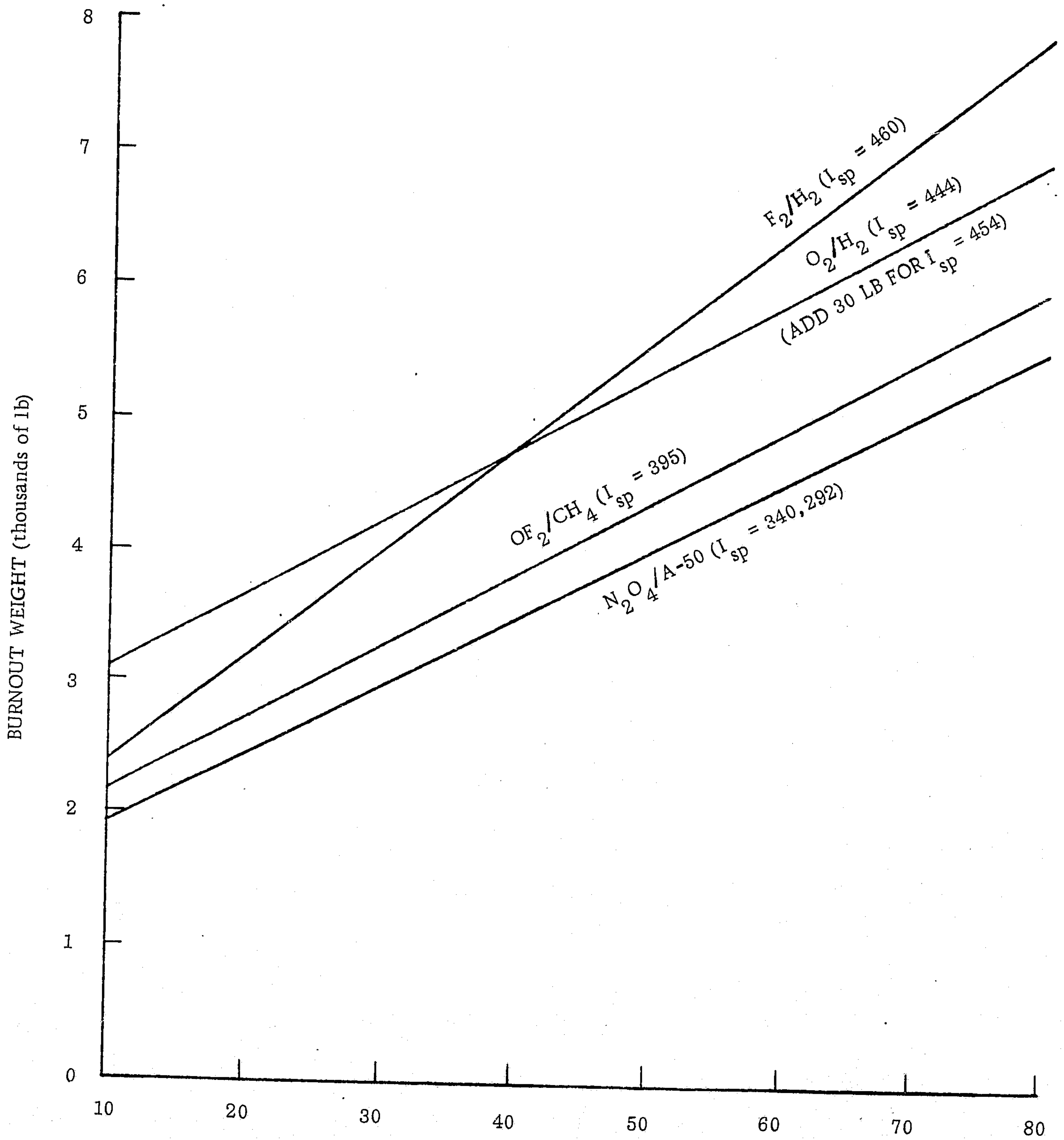


Figure 4-32. Propulsive Stage Scaling Laws

A single design was selected for analysis using a 50,000-lb gross weight single engine O_2H_2 configuration with specific impulse of 444 sec. The weight statement for this configuration (including engine start and stop loss, propellant venting, and attitude control requirements) is presented in Section 4.4.1. The payload for the synchronous equatorial and synchronous inclined orbit is 3,880 and 7,970 lb respectively.

A curve of payload versus excess circular velocity for a single burn mission is presented in Figure 4-33 based on the O_2H_2 point design configuration. The discontinuous break in the curve results at the point at which propellant is off-loaded to keep the gross weight from exceeding 50,000 and 80,000 lb, respectively. As the weights for the specific configurations are developed, the performance curves will be revised.

Table 4-10. Performance Curve Identification

Mission Mode	Operational Mode
Synchronous Equatorial Circular Orbit	Payload placement - expendable vehicle (Figure 4-34)
	Payload placement - recoverable vehicle (Figure 4-35)
	Payload retrieval (Figure 4-36)
	Payload placement and retrieval (Figure 4-37)
Synchronous Inclined Circular Orbit	Payload placement - expendable vehicle (Figure 4-38)
	Payload placement - recoverable vehicle (Figure 4-39)
	Payload retrieval - recoverable vehicle (Figure 4-40)
	Payload placement and retrieval - recoverable vehicle (Figure 4-41)

Payload partials are presented in Table 4-11. These partials are valid only for the conditions for which they were calculated and should not be assumed valid for other conditions.

4.4.5 ORBITAL OPERATIONS. This section discusses the orbital operation of deploying the propulsive stage payload with the space shuttle and docking after the propulsive stage has completed its mission.

4.4.5.1 Propulsive Stage Deployment. A method of extracting the propulsive stage/payload from the cargo bay of the orbiter while in orbit is required. The propulsive stage attitude control system could possibly be used for this purpose, but the space available for maneuvering inside the cargo bay is very limited and exhaust impingement on internal orbiter hardware is not desirable, which makes this approach appear impractical. Some mechanical method of deploying the propulsive stage is therefore needed. Many mechanisms are capable of performing this separation. Among those considered were slider, rotating arm, and four-bar linkage mechanisms. Figure

NOTES:
1. ONE BURN
2. REFERENCE I_{sp} = 444
3. JETTISON WEIGHT = 4820
4. FPR = 0.02% ΔV
5. PROPELLANT WEIGHT = 38,908 (MAXIMUM)

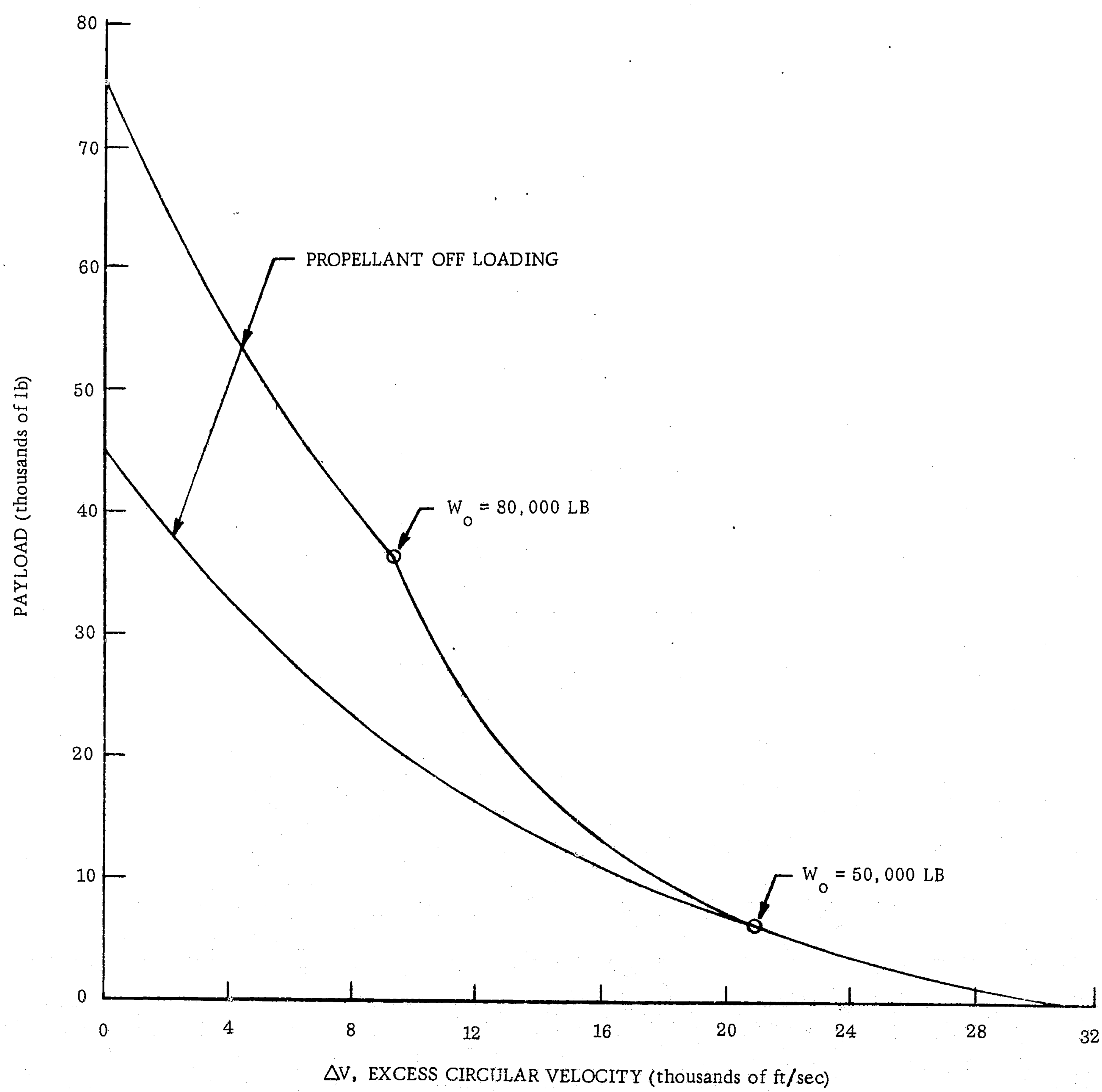


Figure 4-33. Propulsive Stage Performance

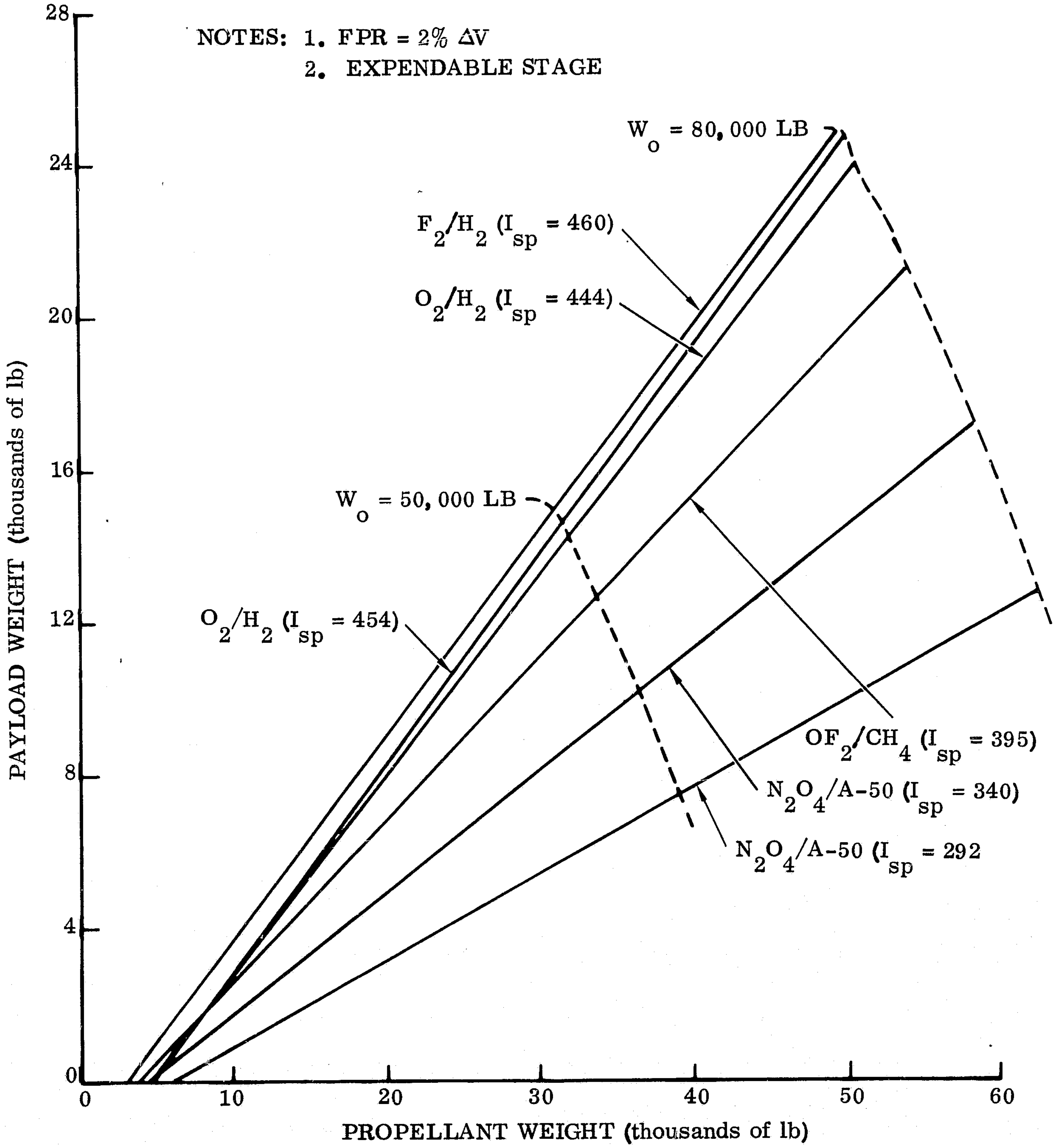


Figure 4-34. Propulsive Stage Performance, Payload Placement in Synchronous Equatorial Orbit

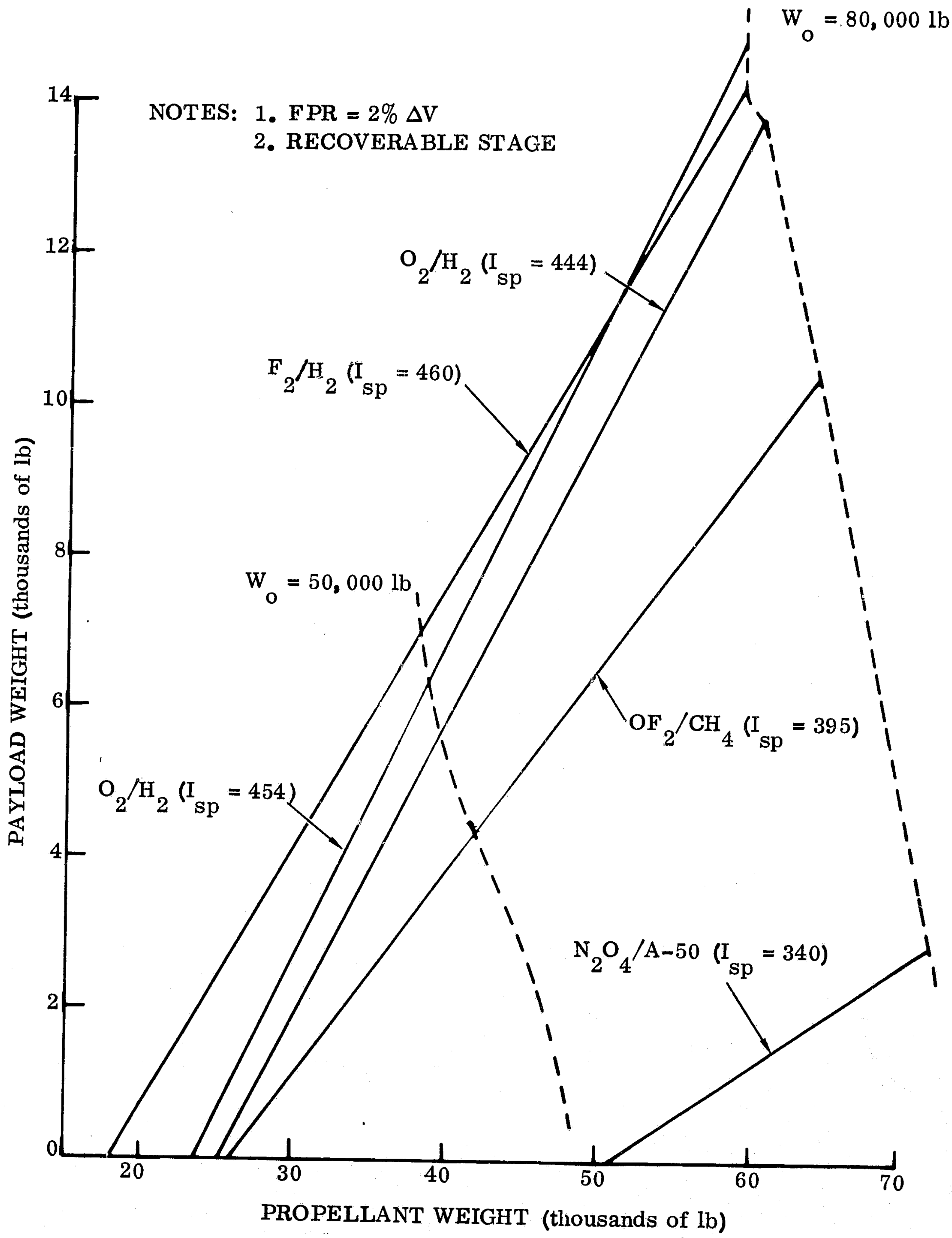


Figure 4-35. Propulsive Stage Performance, Payload Placement in Synchronous Equatorial Orbit

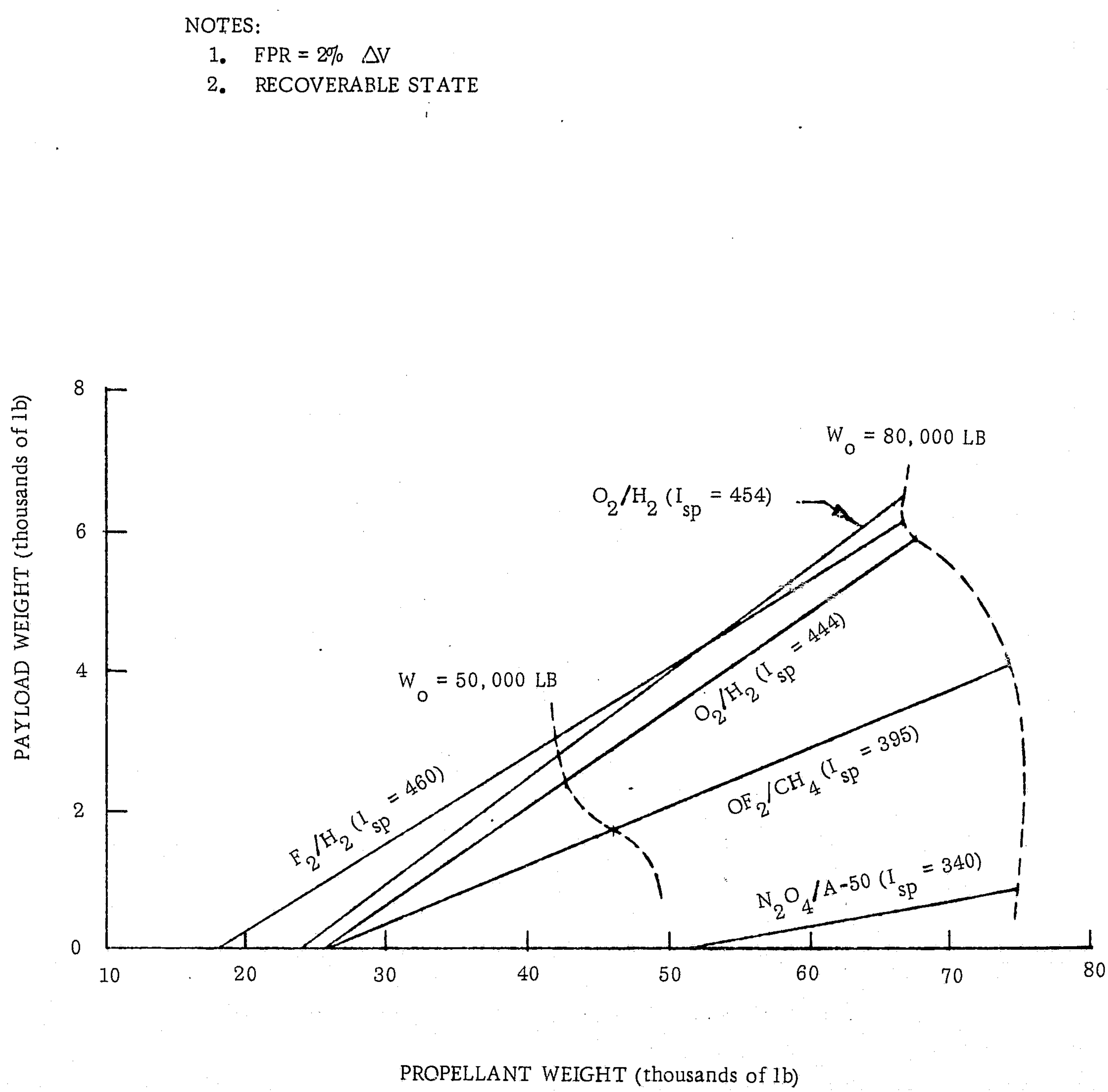


Figure 4-36. Propulsive Stage Performance, Payload Retrieval in Synchronous Equatorial Orbit

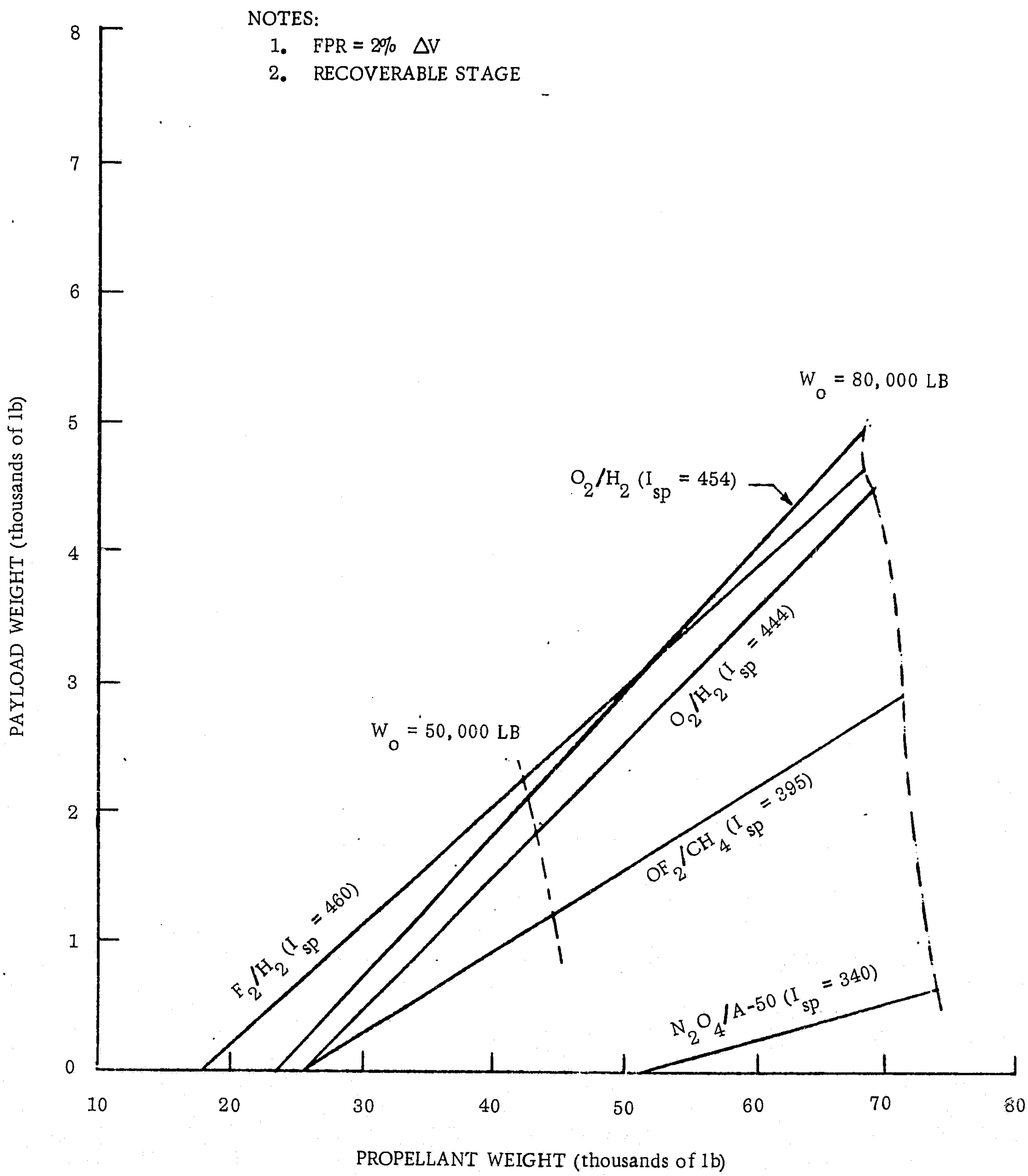


Figure 4-37. Propulsive Stage Performance, Payload Placement and Retrieval in Synchronous Equatorial Orbit

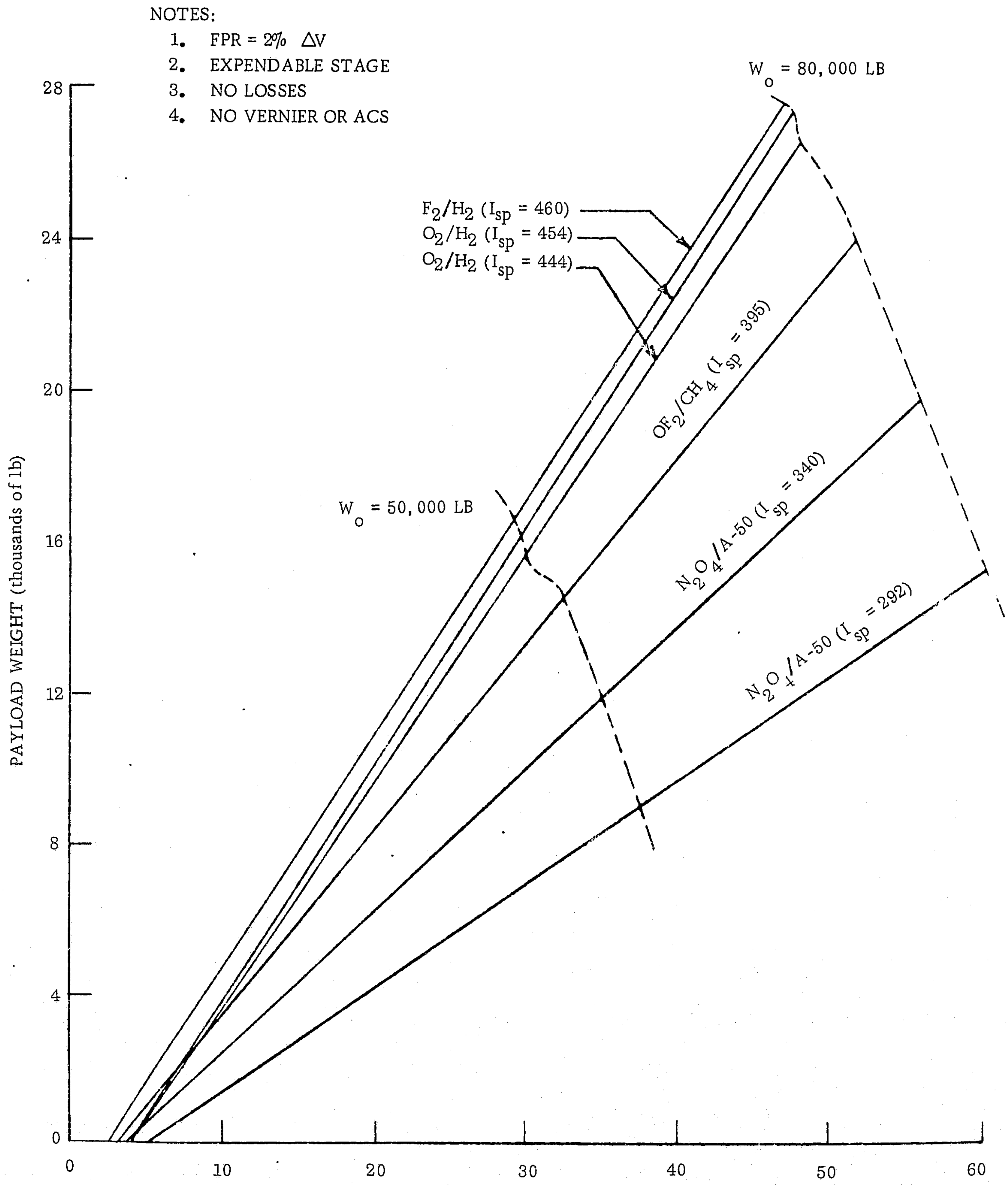


Figure 4-38. Propulsive Stage Performance, Payload Placement in Synchronous Inclined Orbit

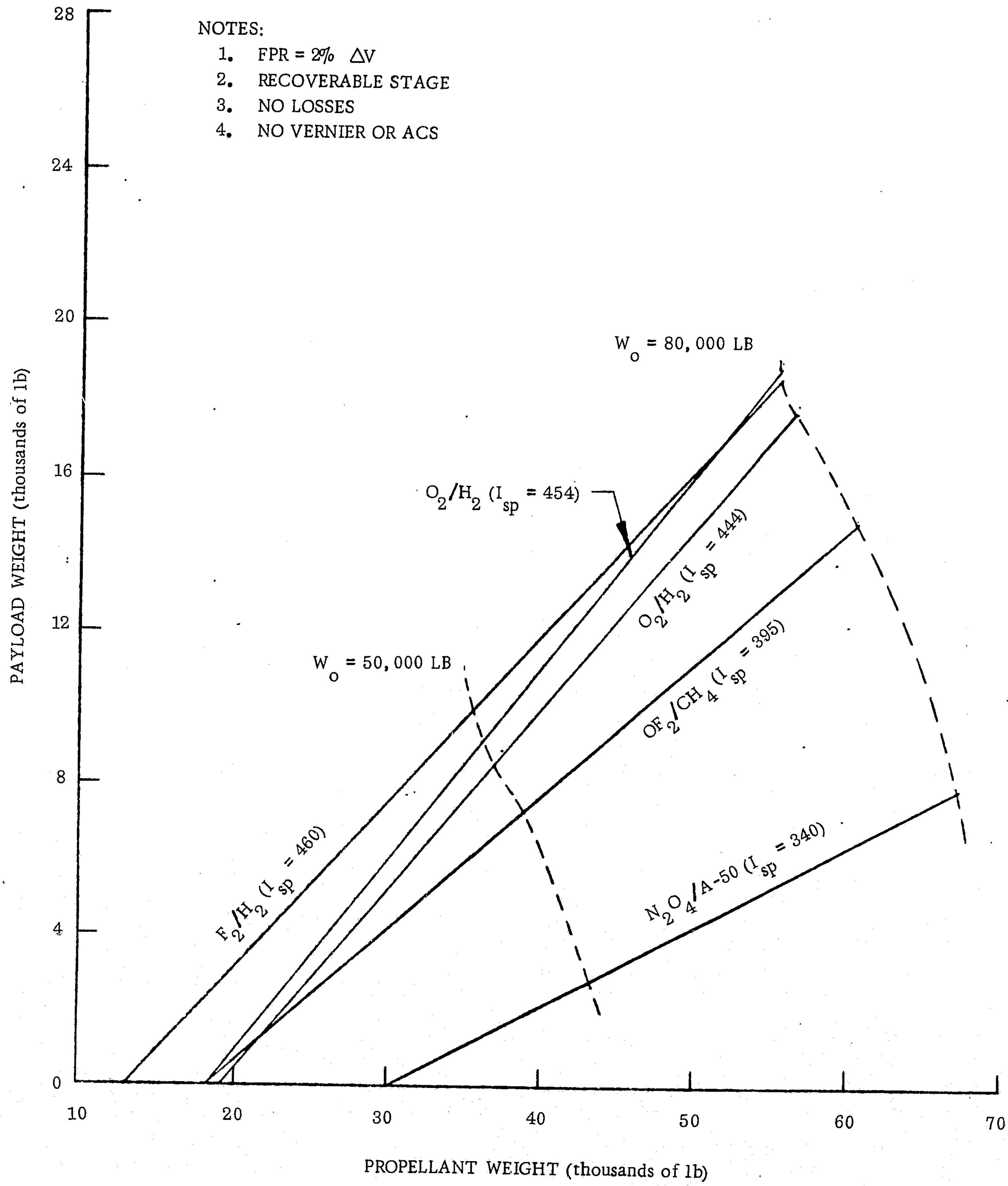


Figure 4-39. Propulsive Stage Performance, Payload Placement in Synchronous Inclined Orbit

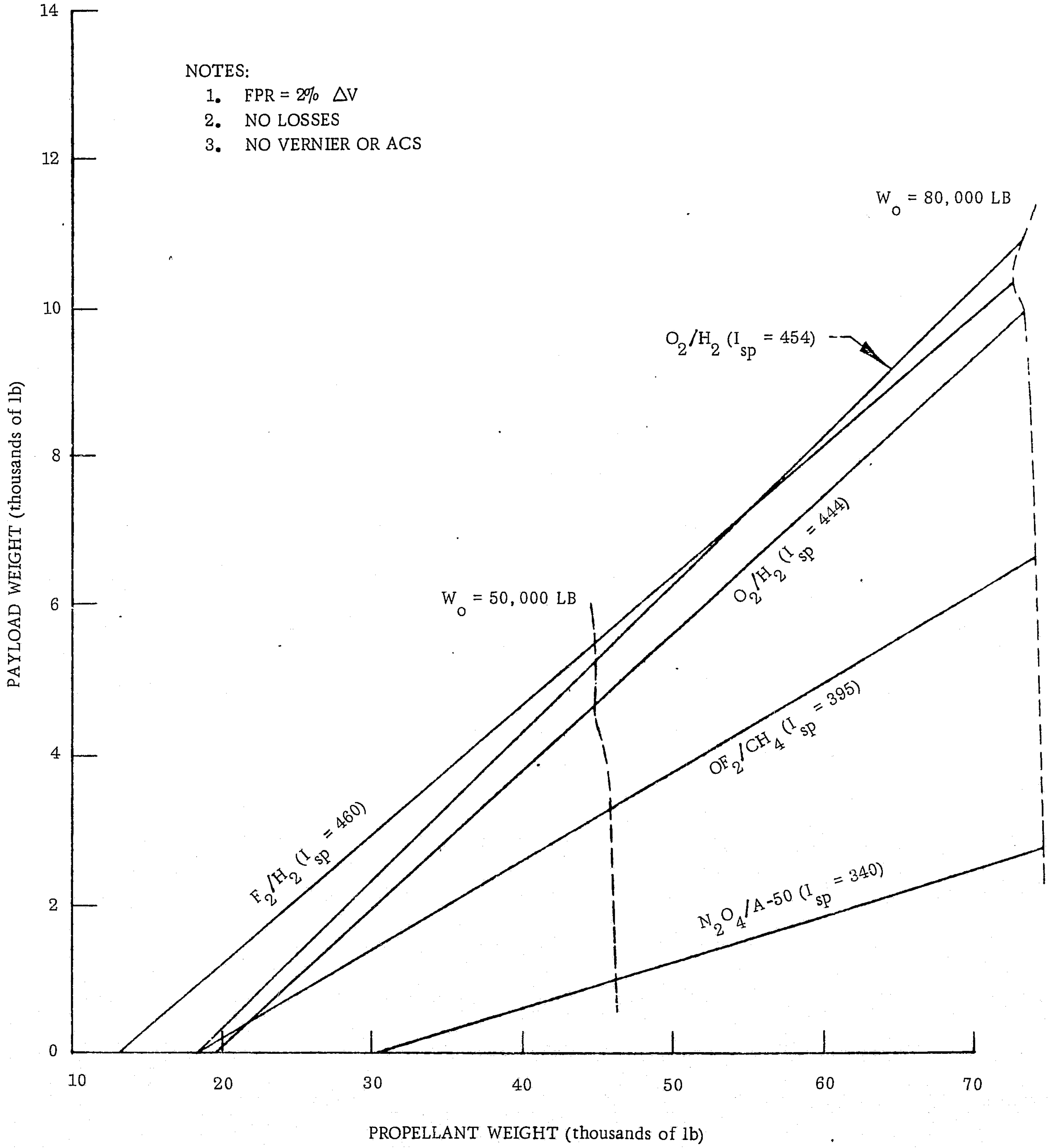


Figure 4-40. Propulsive Stage Performance, Payload Retrieval from Synchronous Inclined Orbit

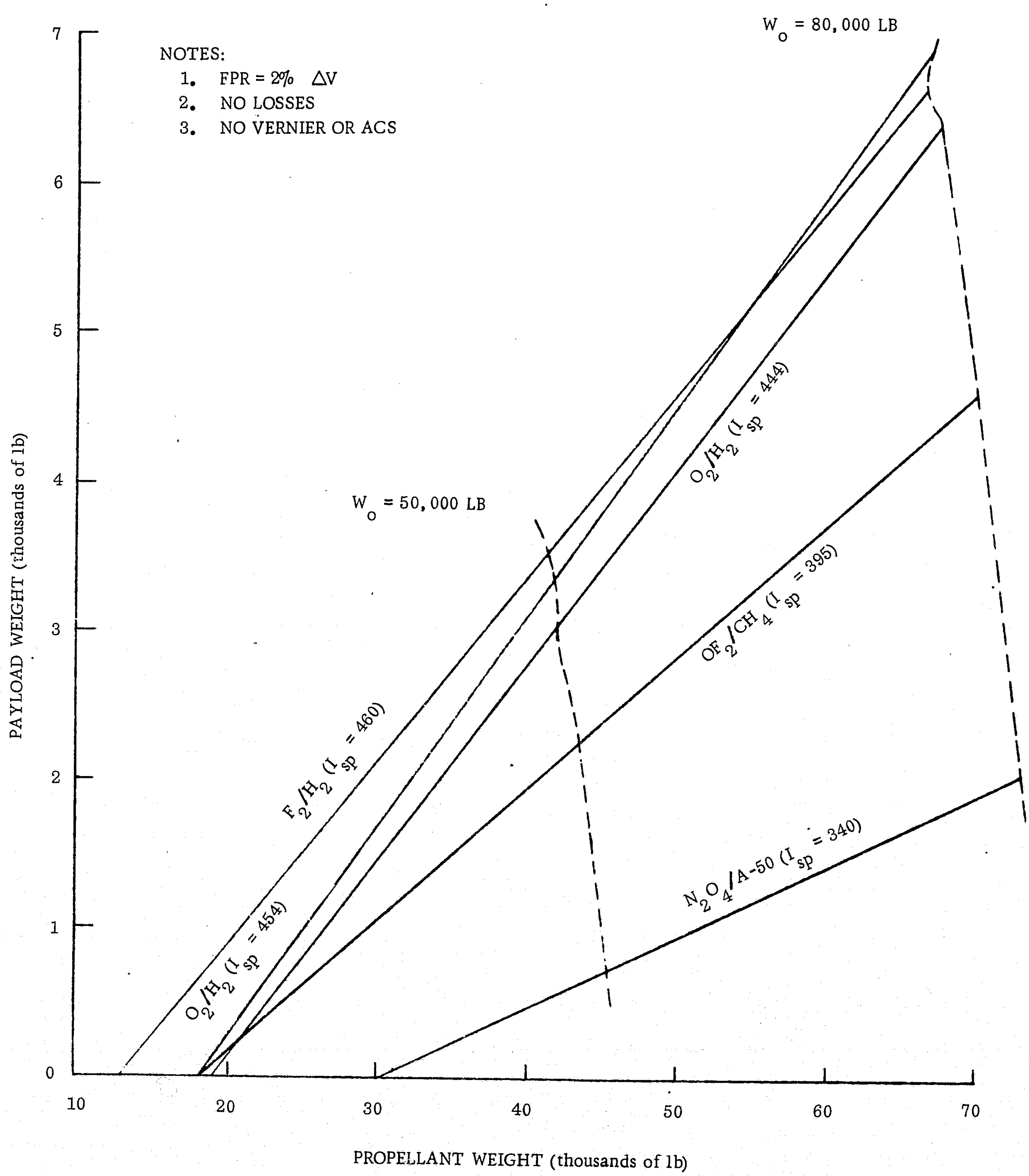


Figure 4-41. Propulsive Stage Performance, Payload Placement and Retrieval in Synchronous Inclined Orbit

Table 4-11. Propulsive Stage Performance Partials

Mission	Payload Mode		Vehicle Mode		Specific Impulse	$\frac{\partial \bar{\omega}}{\partial \omega_j}$	$\frac{\partial \bar{\omega}}{\partial I}$	$\frac{\partial \bar{\omega}}{\partial \omega_p}$	$\frac{\partial \bar{\omega}}{\partial FPR}$
	Placement	Retrieval	Expendable	Recoverable					
Synchronous Equatorial	X			X	444	-3.8	144	0.60	
Synchronous Equatorial	X		X		444	-1.0	62.4	0.58	
Synchronous Equatorial		X		X	444	-1.35	56.6	0.21	
Synchronous Inclined	X			X	444	-3.6	126	0.68	-456
Synchronous Equatorial	X			X	460	-3.74	112	0.63	
Synchronous Equatorial	X			X	395	-4.2	184	0.50	
Synchronous Equatorial	X			X	340	-4.86	219	0.39	

Symbols:

$\bar{\omega}$	payload weight (lb)
ω_j	jettison weight (lb)
ω_p	propellant weight (lb)
I	specific impulse (sec)
FPR	flight performance reserve (percent)

4-30 shows a typical four-bar linkage with a rotating arm follower. The linkage rotation points and drive (either electric ball-screw or pneumatic/hydraulic cylinder) are fixed to the orbiter cargo bay longerons, and the driven points are attached to the external interface support structure. The linkage shown results in an almost linear "straight out" travel of the front of the payload, clears the main structural frames at the fore and aft end of the cargo bay, and completely removes the propulsive stage/ payload from the orbiter cargo bay. Since this linkage operates only in a zero-g environment, the only forces applied are those due to inertial loads. No specific deployment time has been specified, but a gradual egress taking approximately 60 sec will result in very low loads and linkage members with realtively small cross-sections.

The deployment linkage used with an internally supported propulsive stage will be very similar to the linkage just discussed. Instead of being attached to the support structure, however, it will be connected to a ring used to support the propulsive stage docking mechanisms.

4.4.5.2 Propulsive Stage Docking. A universal docking ring is desirable for capture and mating of the orbiter and its payload. This docking ring concept allows either identical half of the device to be active while the other half is passive. It is particularly attractive for use with manned payloads since pressurized personnel tunnels can be coupled through the inside of the ring. For the propulsive stage, no personnel transfer is envisioned, particularly at the aft end. This ring configuration fits well about the engine, however, and uses very little cargo bay space.

The docking ring shown on the propulsive stage, Figure 4-23, is completely passive, while that in the orbiter, Figure 4-30, is the active ring.

In addition to the normal docking requirements of impact absorption, capture, and mating, another special action is needed during the docking sequence. This is rotational positioning. Lateral structural support between the propulsive stage and the orbiter is required during reentry, maneuvering, and landing. These lateral supports are located on the horizontal axis of the propulsive stage and are engaged by support pins attached to the cargo bay longerons. The propulsive stage must be carefully positioned with respect to the orbiter to enable proper pin engagement.

A positioning ring is used to provide both the translational and rotational motion to the docking ring. The positioning ring is constructed of two interconnected concentric rings which are free to turn in their supports and with respect to each other. There are four supports holding the positioning ring; one is a drive unit and the other three are follower units. These support guides are attached to the external support structure by shock mounts to absorb impact loads during docking. Three pairs of support struts connect the positioning ring to the docking ring. Of each pair, the left leg is connected to the outer half of the positioning ring and the right leg to the inner half. Therefore, when the positioning ring halves are counter rotated, an in-and-out translational movement of the docking ring is obtained. When the ring halves are both rotated together as one, the docking ring also rotates but does not translate.

The propulsive stage deployment/docking sequence is:

a. Deployment and Release

1. Demate propulsive stage/orbiter electrical/fluids disconnects.
2. Unlatch axial and lateral structural support attachments.
3. Remove propulsive stage from the orbiter cargo bay with the four-bar deployment mechanism.
4. Deploy propulsive stage attitude control system thruster clusters.
5. Unlatch propulsive stage/external interface support structure structural latches (24).
6. Translate propulsive stage away from the interface support structure with docking positioning mechanism.
7. Unlatch docking ring capture hooks.
8. Move propulsive stage away from the orbiter with ACS engines.

b. Docking and Storage

1. Propulsive stage is positioned slightly ahead of the orbiter in proper docking attitude with the propulsive stage ACS system.
2. Interface support structure is in deployment/recapture position with structural latches released.
3. Docking ring is fully extended with capture hooks open.
4. The propulsive stage is driven to the orbiter using its ACS engines.
5. Positioning for docking is accomplished by aids such as closed circuit television.
6. Capture hooks on the propulsive stage docking ring engage orbiter docking ring and latch.
7. Docking ring is retracted and rotated to proper propulsive stage/orbiter orientation.
8. Propulsive stage is mated with the internal support structure and the structural latches are engaged.
9. The ACS engine clusters are retracted, nitrogen purge connected, and propellant tanks blown down and refilled with GN_2.
10. The propulsive stage is retracted into the cargo bay by the deployment linkage.
11. The axial and lateral structural support attachments are engaged.

4.4.6 GROUND OPERATIONS. Propulsive stages using solid engines and/or storable propellants could be installed in the orbiter payload section at the logistics area as illustrated in Figure 4-42. The orbiter element will be towed to this area in the horizontal mode, the cargo doors opened, and the stage installed by means of a loading device similar to that shown in the figure. The orbiter then is towed to the launch pad.

When the propulsion stage contains cryogenic propellants, such as the baseline defined in this section, the stage is installed dry at the logistics area and propellant added at the launch pad. This servicing requirement was discussed in Section 4.4.3.1, Fluid System Interfaces.

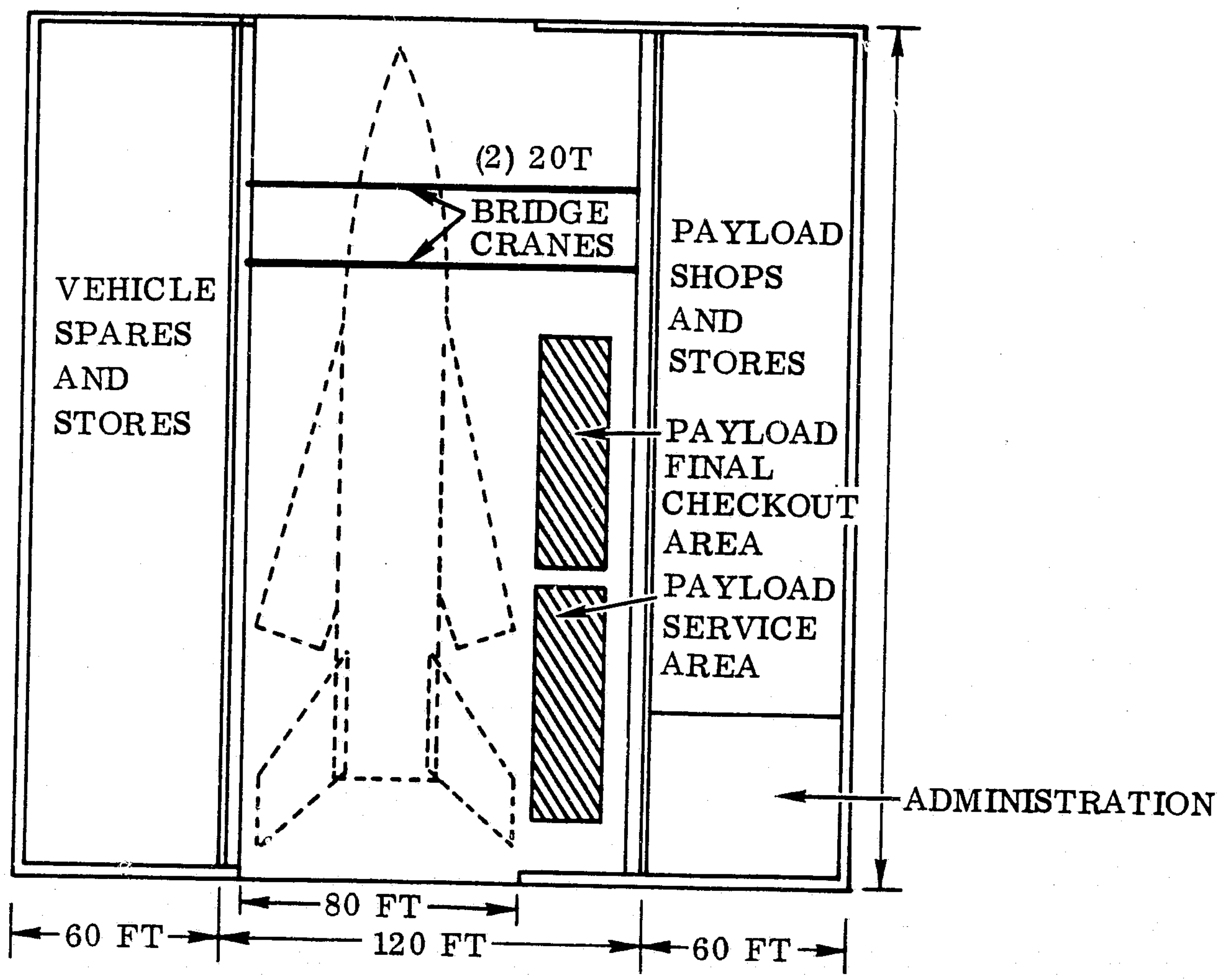


Figure 4-42. Payload Installation

4.4.7 MULTI-MISSION CAPABILITY. This payload extends the mission flexibility of the space shuttle basically by adding another stage to the orbiter.

The baseline payload vehicle described in this section, for example, can place payloads in synchronous equatorial or inclined circular orbit. In addition, the system has the following operational modes available:

a. Payload placement - expendable propulsive stage

b. Payload placement - recoverable propulsive stage

c. Payload retrieval (from synchronous orbit)

d. Payload placement and retrieval

The performance (payload weight, etc.) on both missions, in the alternate modes of operation is shown in Section 4.4.4.

4.5 SATELLITE DELIVERY, MAINTENANCE, AND RETRIEVAL

This section documents a brief study to determine space shuttle requirements resulting from the satellite mission. Alternate maintenance and deployment concepts are shown as well as a method of retrieving an inoperative unmanned satellite.

4.5.1 BASIC MAINTENANCE CONCEPTS. As shown in Figure 4-43, several concepts were considered for servicing a satellite in Earth orbit.

Concept A would retrieve the inoperative satellite, return it to Earth for maintenance, and then redeploy it in orbit. This approach eliminates the problems associated with performing maintenance in orbit, such as pressurized work areas and having equipment available. However, two space shuttle round trips are required and the costs involved make this an unattractive concept. This approach could be used with minimum modification to the space shuttle after the addition of a docking adapter (provided the satellite had docking provisions).

Concept B assumes that the satellites deployed during the time period of interest have in-orbit maintenance provisions incorporated. Such provisions would include a pressurized work section containing components most likely to fail and a docking port. In this case, the maintenance personnel and equipment could enter the satellite after docking. This concept would also minimize special space shuttle requirements. The primary disadvantage is that this concept requires all satellites have maintenance provisions incorporated.

Concept C requires a pressurized payload bay to permit maintenance in a shirtsleeve environment. It was estimated that this would result in a 15,000-lb penalty to the basic space shuttle orbiter on all missions not requiring a pressurized payload bay.

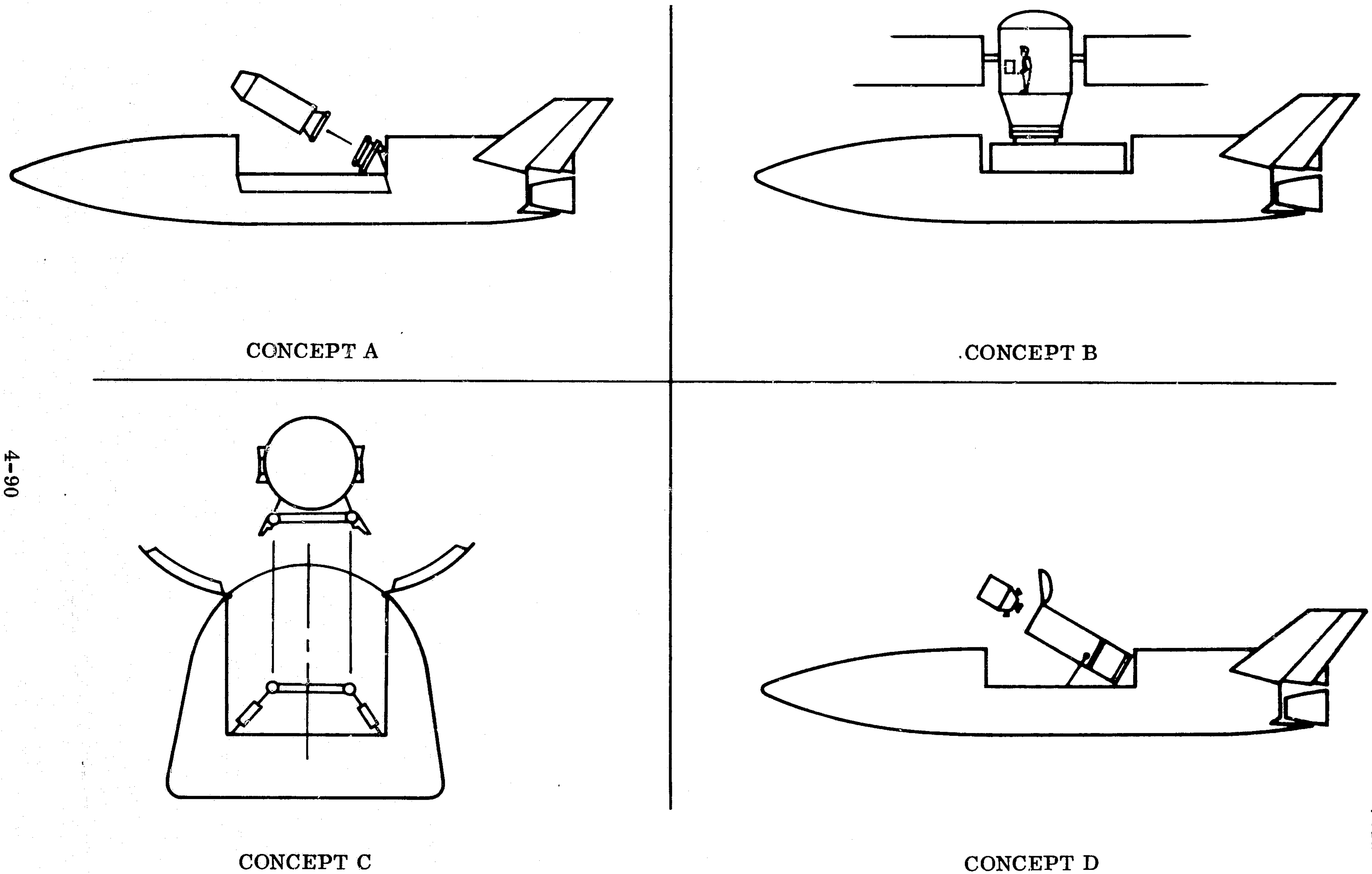


Figure 4-43. Basic Satellite Maintenance Concepts

Concept D uses a pressurized maintenance module in the space shuttle. The module is installed in the payload bay and rotated to the position shown. The end of the module opens and the satellite is maneuvered into the module. The end then closes and the module pressurized. Workmen enter from the small personnel module. This concept would not unduly penalize the satellite or the basic space shuttle and is recommended as a baseline concept.

4.5.2 DEPLOYMENT AND RETRIEVAL CONCEPTS. Satellite deployment is a considerably easier task to accomplish than retrieval because the payload is directly connected to the orbiter throughout the deployment phase giving absolute control of the separating bodies. Retrieval presents the most formidable problems particularly when encountering a random tumbling,rolling payload of the size and weight that the orbiter is designed to carry. The following discussion, therefore, is directed primarily towards the retrieval operation.

Two satellite retrieval docking concepts were analyzed: non-impact and impact. The non-impact concept is performed with a winch drawing the two bodies together against the resisting force of an RCS at low closing velocities; the impact concept employs a "double ring and cone" docking system with shock absorbers. (See Figure 4-43.)

With the large inertia loads and satellite bodies of relatively lightweight structure it is apparent that there is a somewhat greater risk to impact dock than to soft dock. The winch/RCS docking system is considered a non-impact docking system since there can be very accurate control of the closing rate between the two bodies. The impact docking is considered to be a standard double ring and cone where hookup would be analogous to freight car couplers with the impact energy being absorbed by the coupling device as discussed in Section 4.2.3.

4.5.2.1 Non-Impact Docking Concept. This concept is based on the capability of stabilizing the payload and orbiter at some distance from each other. A structural member with a direction detector such as TV will then be extended from the orbiter to engage the satellite. The light structural member is slowly retracted to bring the vehicles together for structural connection.

The extendable member, a boom tape, is capable of transmitting tension loads primarily and only a minor amount of compression. Compression would be used to extend the member for attachment to the payload at a distance of about 20 feet.

The winch would accelerate the bodies toward each other with retro-thrust on both the payload and orbiter providing counterforce. The non-impact docking concept depends on achieving very low relative velocities between the two bodies. This is in comparison with the current Apollo impact conditions. Winches in general provide slow smooth positive pull and are ideal for meeting the docking requirements.

Figure 4-44 shows schematically a non-impact docking operation. The sequence of events required to accomplish this mission is described in the next paragraphs. A time graph, Figure 4-45, shows a two-minute non-impact docking operation. This graph assumes the payload has maneuvered 20 feet from the orbiter's deployment/ retrieval (D/R) mechanism and the winch boom is connected to the payload. The final docking sequence begins at this point.

The graph shows that the payload is accelerated by the winch pull against the small retro-thrust (50 lb) of the orbiter's and payload's stabilization systems. The winch pulls at 0.25 fps until the payload is three feet above the D/R mechanism. At this point either the retro-thrust is applied which exceeds the winch's slip clutch capacity or the retro-thrust remains the same and a variable speed winch slows down as the payload decelerates. This depends on the winch concept as discussed.

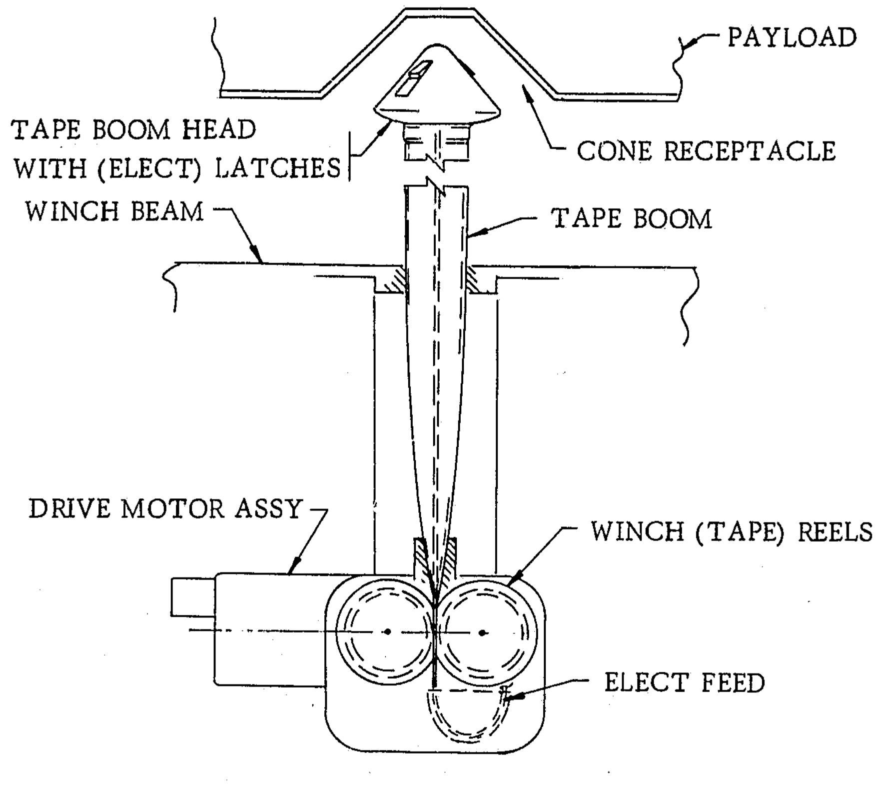
After payload deceleration, the payload is two feet above the D/R mechanism and possesses a closing velocity of 0.05 fps. At touchdown the closing rate is reduced to zero and the payload is at rest on the D/R mechanism. The payload is then checked to see if automatic latching has been successfully achieved before releasing winch and deactivating the ACS system. Propellant required by the payload's ACS system is shown in Figure 4-46.

Once the payload is on the telescoping D/R mechanism, it is very slowly retracted into the payload bay where it is latched to the orbiter's structure as described in the next section.

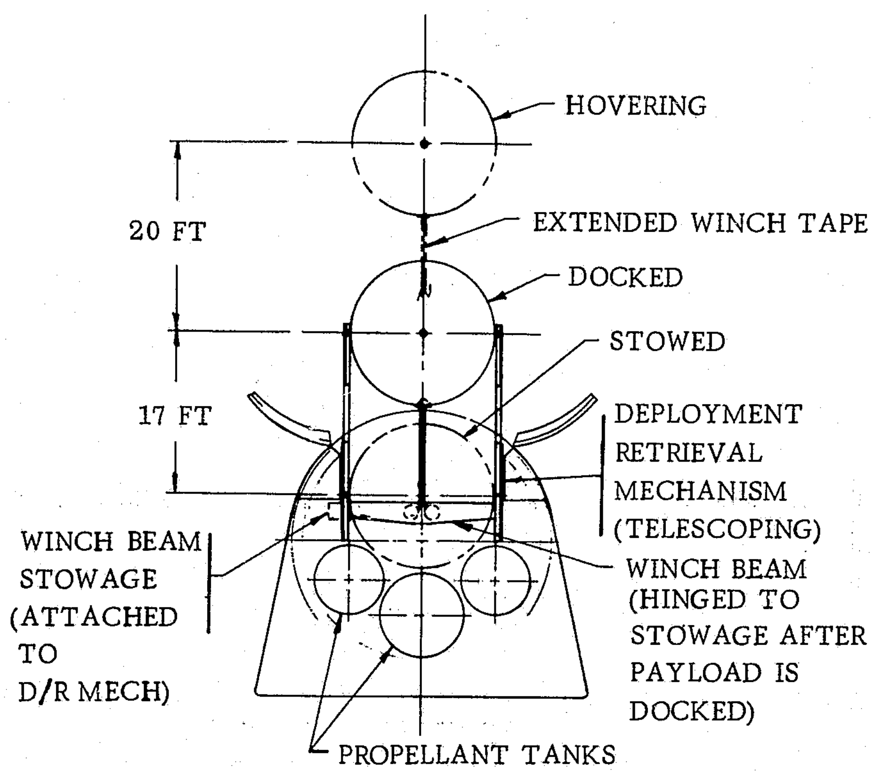
The D/R mechanism is mounted on a track that rides on rails along the inboard side of the payload bay longerons. This allows adjustment to accommodate various c.g. positions of the payload and placement of several smaller payloads in the same bay. The retraction winch is mounted on an under carriage or on its own track to align with the payload's retrieval cone receptacle when its lugs are lined up with the D/R mechanism receptacles.

Two types of winch concepts have been considered: (1) a constant speed reel with a slip clutch, which during the docking operation slips against the decelerating thrust of the payload docking pack RSC, and (2) a variable speed reel which slows down as the payload is decelerating and which also has slip clutch capability during acceleration of the payload.

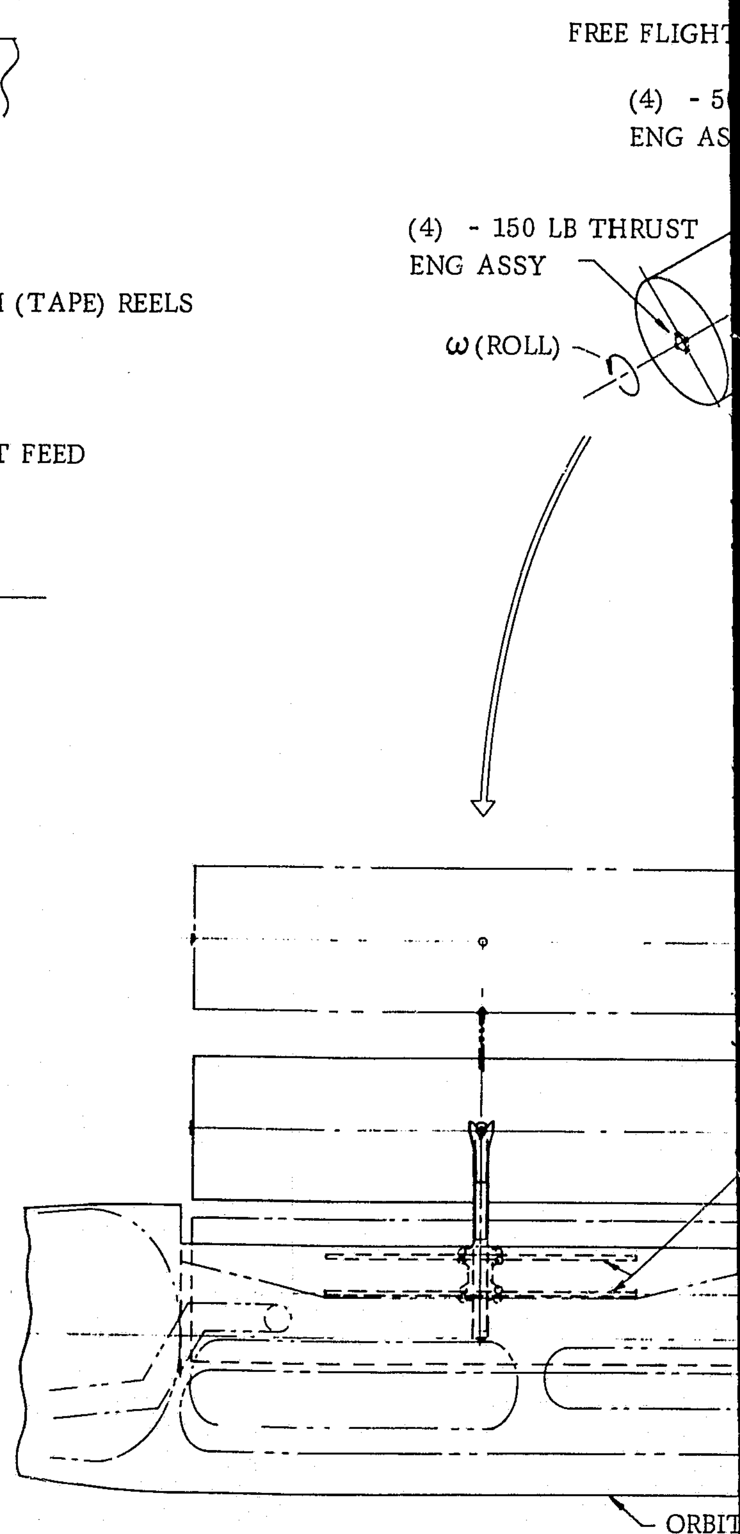
The constant speed reel provides a positive pull at all times. This prevents the possibility of buckling the winch tape column and ruining it for future operations. The variable speed reel may inadvertently be set to reel-in slower than the payload's decelerating velocity and buckle the column. However, the variable speed winch operation requires about half the propellant from the docking pack that the constant speed winch operation consumes (21 lb to 55 lb). This means that doing the same job over the same time more loss work is dissipated as heat energy by the constant speed winch operation.



WINCH ASSY



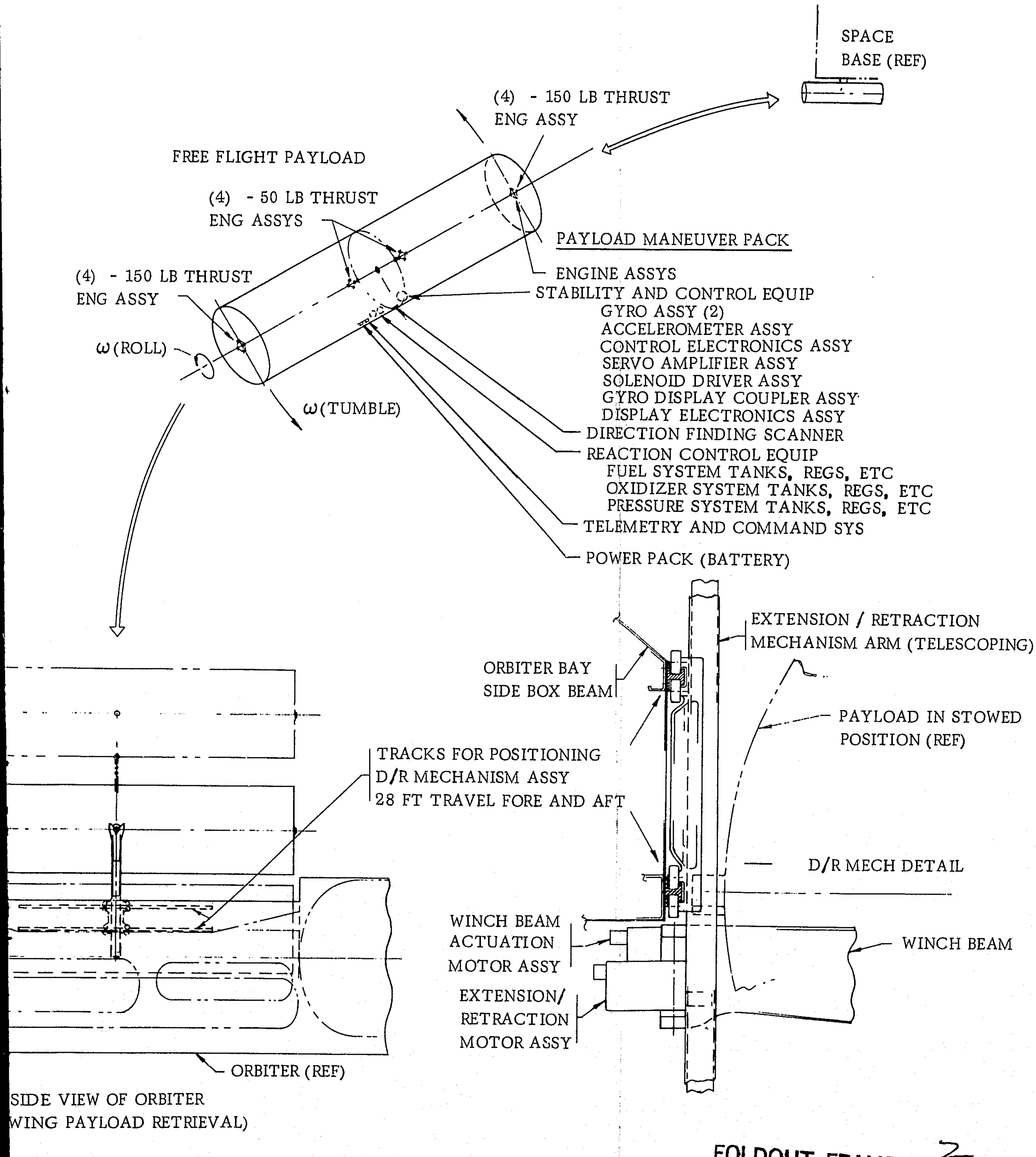
SECTION



SIDE VIEW OF ORBITER
(SHOWING PAYLOAD RETRIEVAL)

FOLDOUT FRAME /

Fi



FOLDOUT FRAME 2

Figure 4-44. Non-Impact Docking System Concept Utilizing a Winch and Maneuver Pads

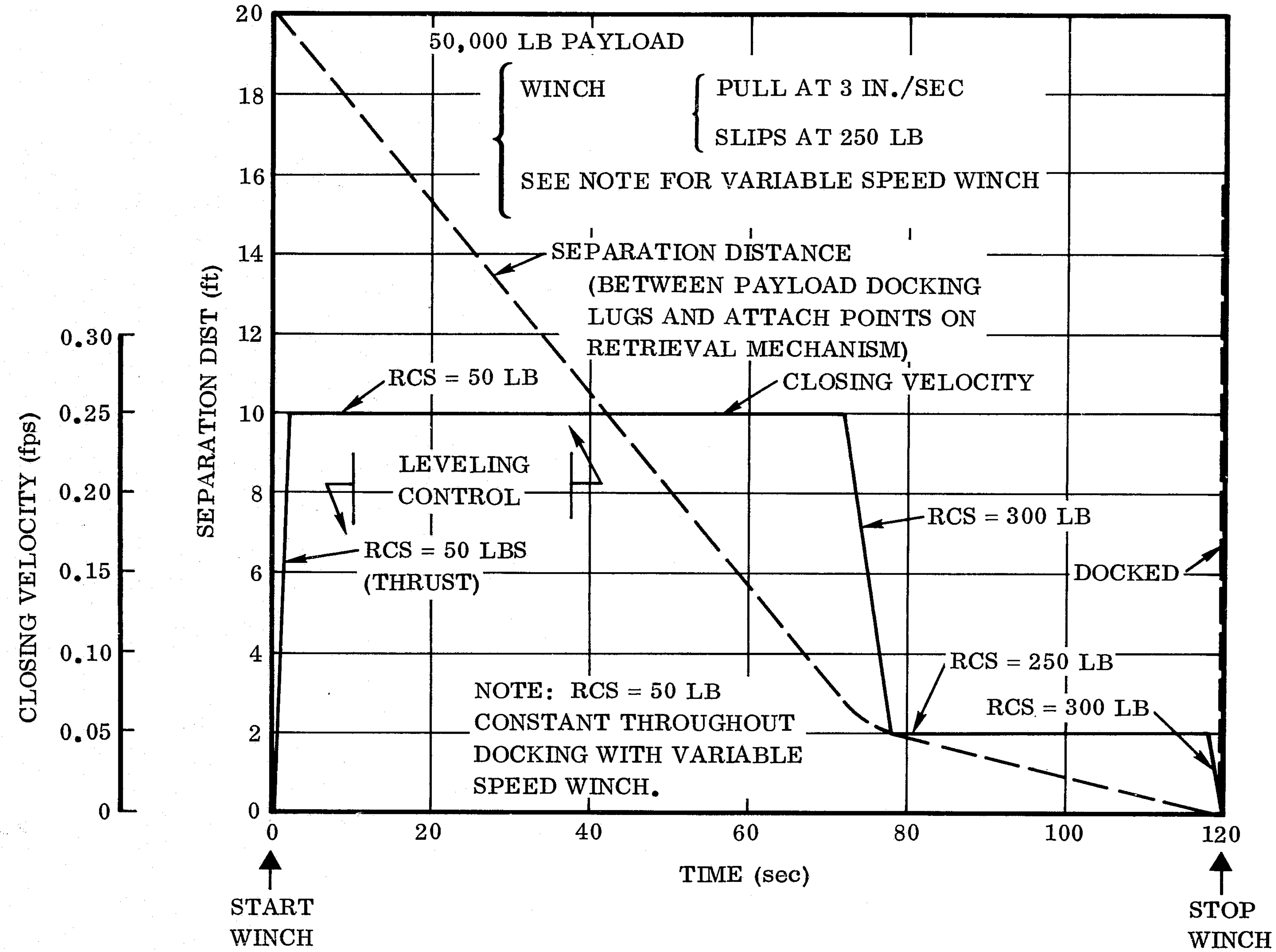


Figure 4-45. Docking Time Velocity and Distance

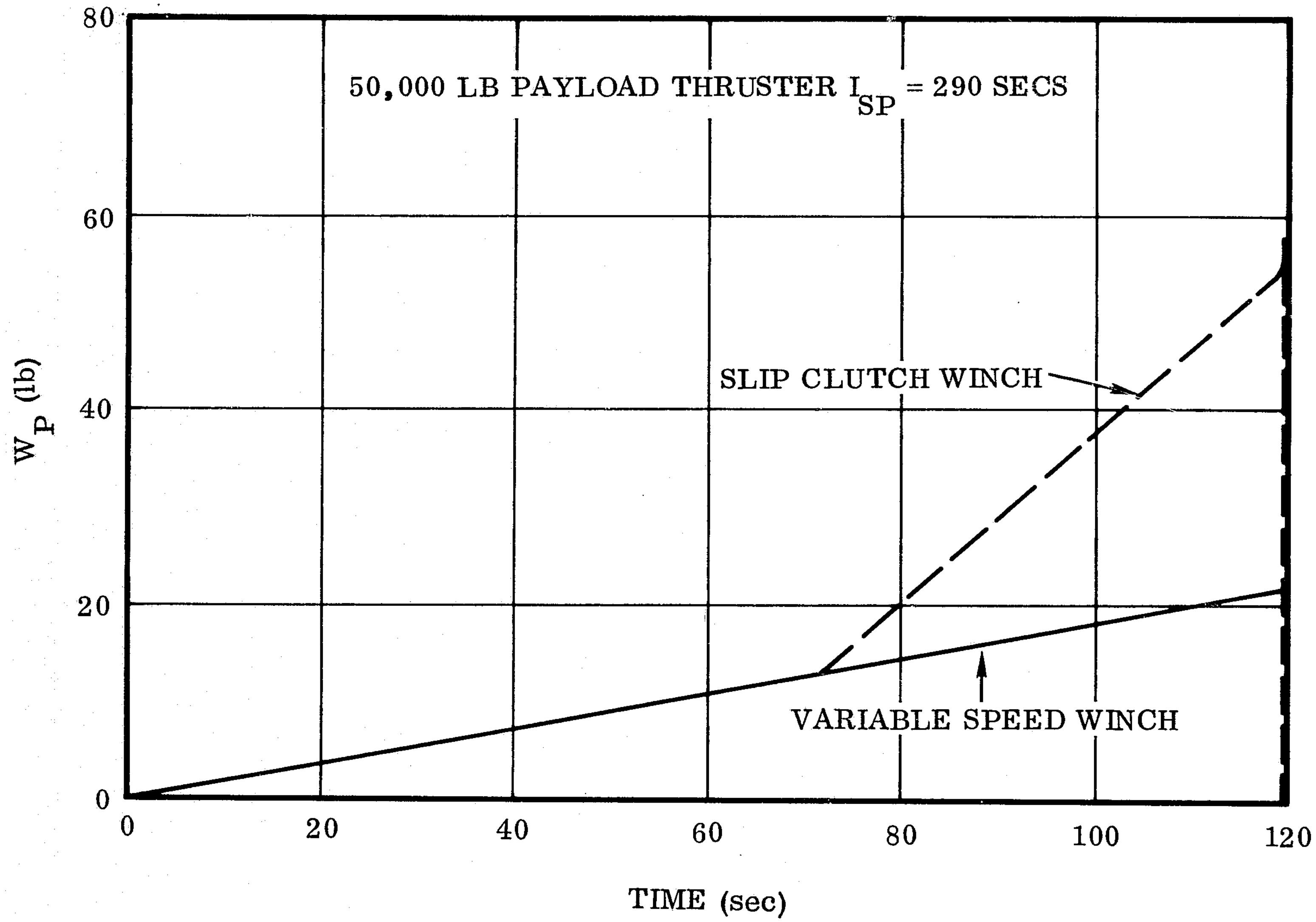


Figure 4-46. Propellant Expended by Maneuver Pack Docking Preflight Payload

The variable speed winch requires no thruster adjustment during docking operation. The constant speed winch varies the thruster output to control docking operation but does not require winch control.

Although there are advantages to both types of winches it appears at present that the variable speed winch has more advantages than the constant speed. The speed change must be smooth to eliminate the chance of negative loads occurring in the tape boom.

In order to accomplish the retrieval task the following list of equipment in the satellite or in the orbiter is necessary:

ACS or maneuver pack (satellite)
Cone receptacle (satellite)
Winch system (orbiter)
Tie down receptacles and chocks (orbiter)
Deployment/retrieval mechanism (orbiter)
Control station (orbiter)

4.5.2.2 Impact Docking Concept. The two mating bodies are positioned and the pilot slowly flys the orbiter to connect with the satellite. This technique is Concept A shown in Figure 4-43. In this concept a payload adapter is employed which is attached to the aft end of the satellite using a standard docking system mechanism. The adapter would be the same diameter as the payload and would include a mating docking system with periphery latching to match the payload. Various mechanisms can be attached to the adapter to translate the satellite module in and out of the payload storage bay. Since the payload is at zero-g during this operation a low-speed screw jack can be used for actuation. Launch and landing loads will be taken by fittings attached to the main longerons.

The payload is positioned at an angle above the orbital vehicle to provide for maximum clearance during docking maneuvers. It is assumed that the payload will be maneuvered into this docking position by either its own ACS or the maneuver pack system described in Section 4.5.3.

The sketches show the adapter and mechanism at the aft end of the payload bay; however, it could be reversed if desired. The rationale behind this concept is to position the payload being retrieved out ahead of the vehicle where it can be seen by the pilot who can then "fly" the orbital vehicle under the payload (analogous to the flying boom and receptacle in airplane refueling operations). If, however, the adapter is located in the forward bay, the payload being retrieved would be behind the orbital vehicle which would seriously hinder the pilot's view and also enhance collisions with the tail surfaces.

The final docking maneuvers would be monitored by a member of the crew stationed in the payload bay area. He would operate the mechanism for latching the payload to the adapter and for translating the payload into the bay. He would then check that the payload has been securely locked into the structural support fittings.

4.5.3 MANEUVERING PACK (MP). A general philosophy that has a great influence on deployment and retrieval methods is investigated herein for the handling of free-flight recoverable satellites. This philosophy postulates that all payloads to be retrieved must possess an active on-board maneuvering system. This system is able to perform all the tasks of a space tug with the additional, most important advantage that it can stabilize a random tumbling and wobbling payload. It is believed that many satellites to be retrieved will have motion that is nearly impossible to control without an internal system. Besides having the ability to stabilize the payload for retrieval, the MP also can translate the payload to the orbiter and assist in the docking operations.

The payload MP is to be in addition to and independent of the operational system allowing it to be activated when the payload is malfunctioning. Generally, the payload will have its operational ACS. This ACS may be inadequate in size or may be the malfunctioning system.

The MP would be remote-controlled by an operator in the orbiter. A link to the ground station would be used to activate the system and check it out before the recovery mission is initiated.

The MP, considered as part of the payload, would consist of the following equipment:

Stability and control equipment
Command and telemetry subsystem
Power supply
Reaction control subsystem and tankage

An alternative to a MP is having the orbiter provide all motion to close-in on the payload. This system may be practical as long as the payload is motionless but the probability is that the inactive payload in free flight will not be motionless but may be tumbling or rolling or the combination which would produce a wobble. This could easily be a result of a malfunctioning stability system which could impart erratic motion to the orbiting payload.

One method of approaching a random tumbling body would be to send up a capture vehicle which would entangle the payload and itself by means of a net, then with the capture vehicle's ACS, the erratic motion could be damped out. Another method would be to secure the tumbling payload with grappling hooks mounted on the rotating end of a pole, the rotational rate of the grappling hook assembly being the same as the payload's rate. This system has a serious limitation where the payload is rolling and tumbling at the same time which would more than likely be the case unless considerable time has elapsed.

An additional vehicle required to perform an arresting task or to maneuver the payload around in space is bound to be heavier and more complex than a built-in system that can perform the identical task. More than likely, the built-in system is more flexible than a separate craft in arresting and maneuvering the payload and can perform its tasks far more quickly with less propellant expended and without the risk of collision.

A highly reliable present state-of-the-art MP system appears more economical to develop than a separate entire vehicle. It also has little or no maintenance or service requirements. It is flexible in its ability to be adapted as a standard unit to all shapes and payload forms.

With large future payloads being delivered to orbit, the basic weight of the MP to the total weight of the payload becomes an increasingly smaller portion of the payload weight. The dry weight is 320 pounds. Loaded with 300 pounds of usable propellant increases the weight to 620 pounds.

Space allocation in the orbiter is of prime importance. A separate craft requires a parking space. Therefore the payload no longer can be the nominal 60-ft long cylinder but must be shortened to accommodate the additional craft volume. The MP on the other hand is distributed throughout the payload and requires only a small portion of its volume.

The maneuver jets with the MP system are located efficiently about the payload's c.g. A separate craft would have all its maneuver jets located at one end of the payload which results in less control and far more expenditure of propellant.

Since the maneuvering pack is used for stabilizing and translating free flight payloads operation time and amount of propellant expended were studied. Figure 4-47 shows the amount of propellant required to stop the motion of the payload about its pitch and roll axes. The payload assumed is a 15 ft diameter by 60 ft long, 50,000-pound cylinder. Its mass is assumed homogeneous.

Four banks of four thrusters each are assumed as standard with 150-pound-thrust jets at the cylinder ends to react tumble and 50-pound-thrust jets near the c.g. to react roll. Further, the storable propellant burned in the thrusters provides a specific impulse of 290 seconds.

It can be seen that considerable amount of energy is required to stop a payload of this size from tumbling or rolling. Figure 4-48 shows the time required to perform the stabilizing task. In addition to the use of 150-pound-thrust units to arrest tumbling, the time required using 100 and 200 lb units is shown in comparison.

Figure 4-49 shows the amount of propellant required using the maneuvering pack to move a 50,000 pound payload from one position in space to another. The times included acceleration and deceleration time. The minimum time limit, shown by the

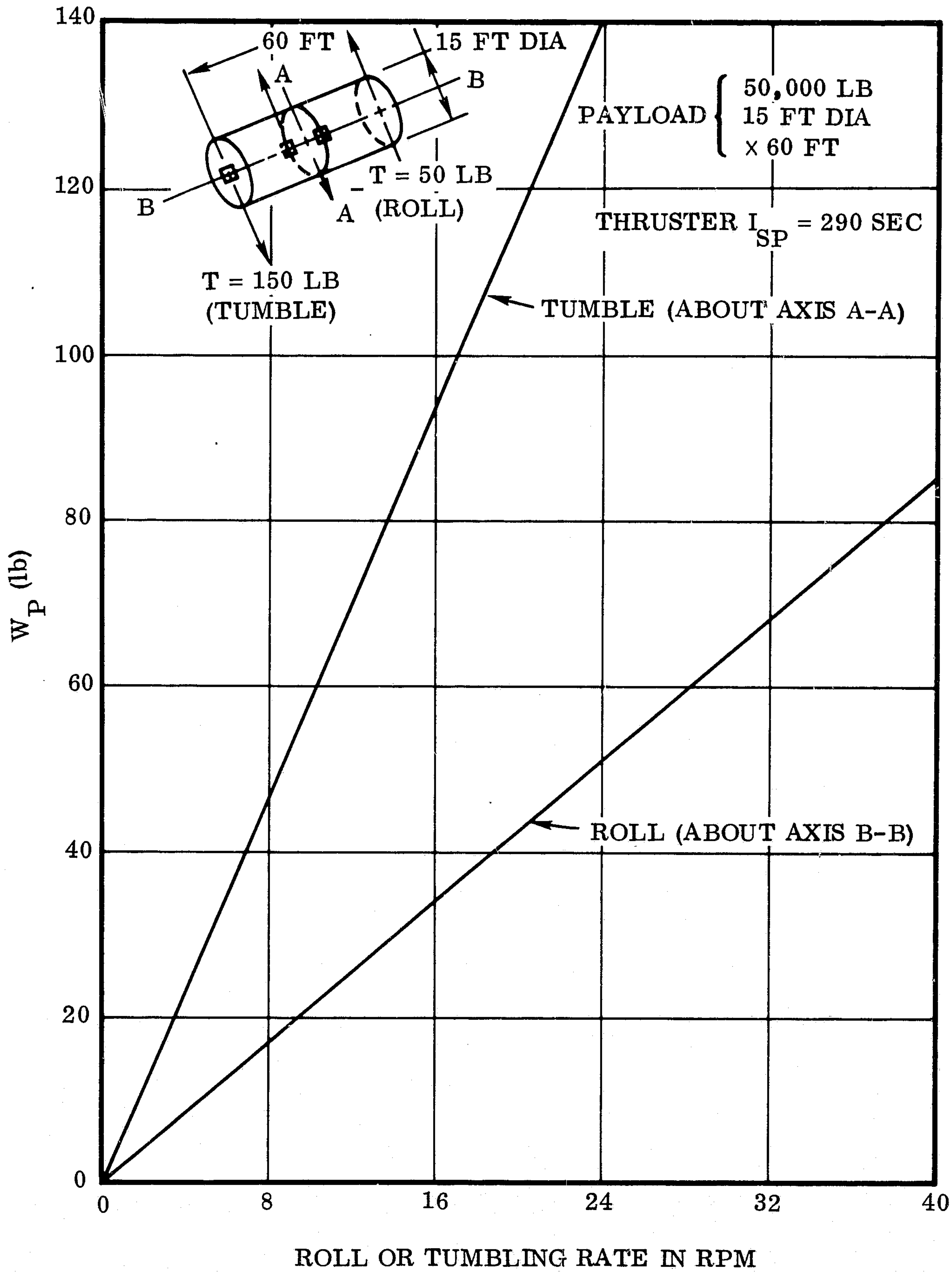


Figure 4-47. Propellant Expended by Maneuver Pack to Stabilize Free Flight Payload

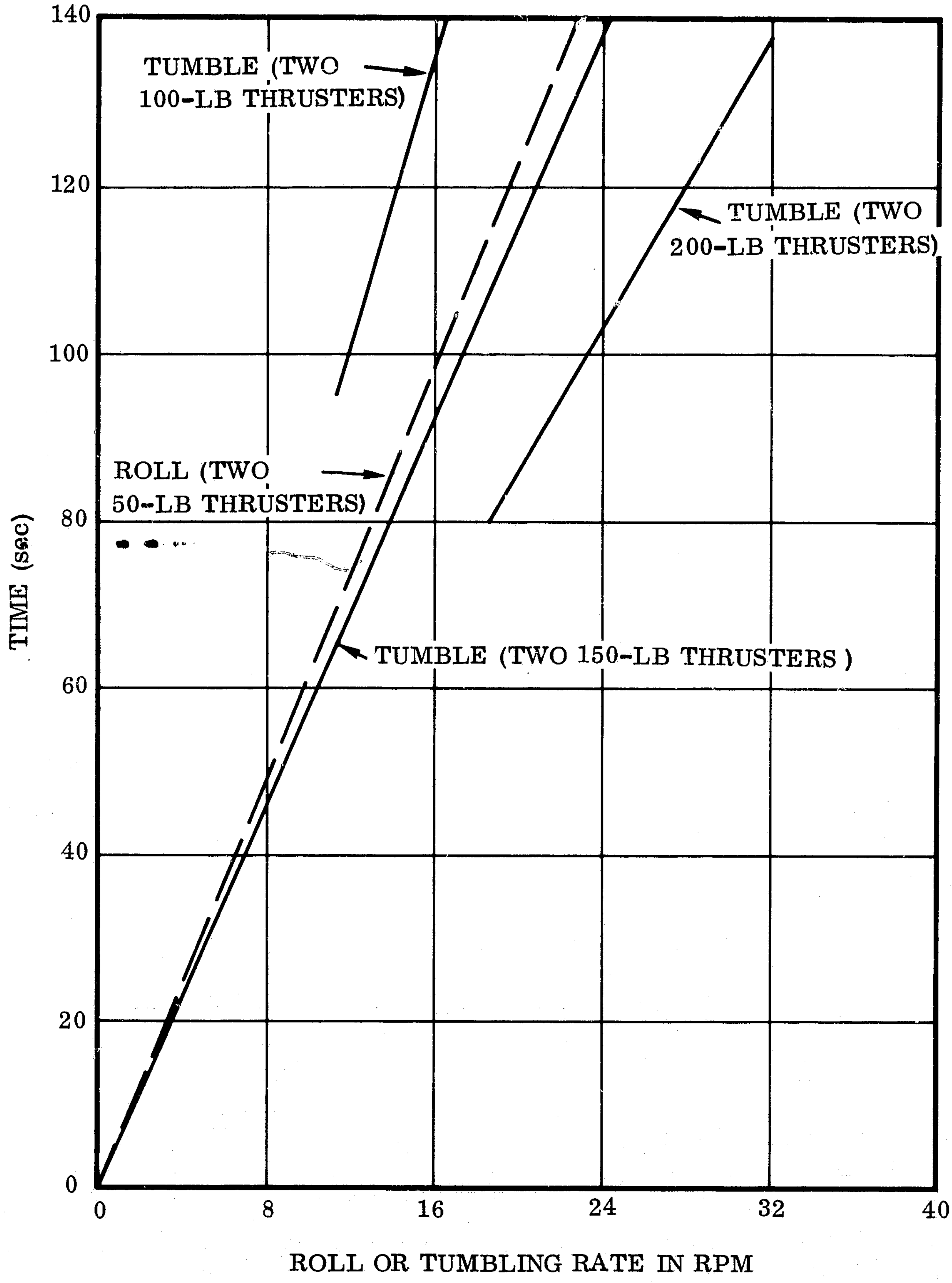


Figure 4-48. Time Required to Stabilize Free Flight Payload with Maneuver Pack; 50,000 lb Payload

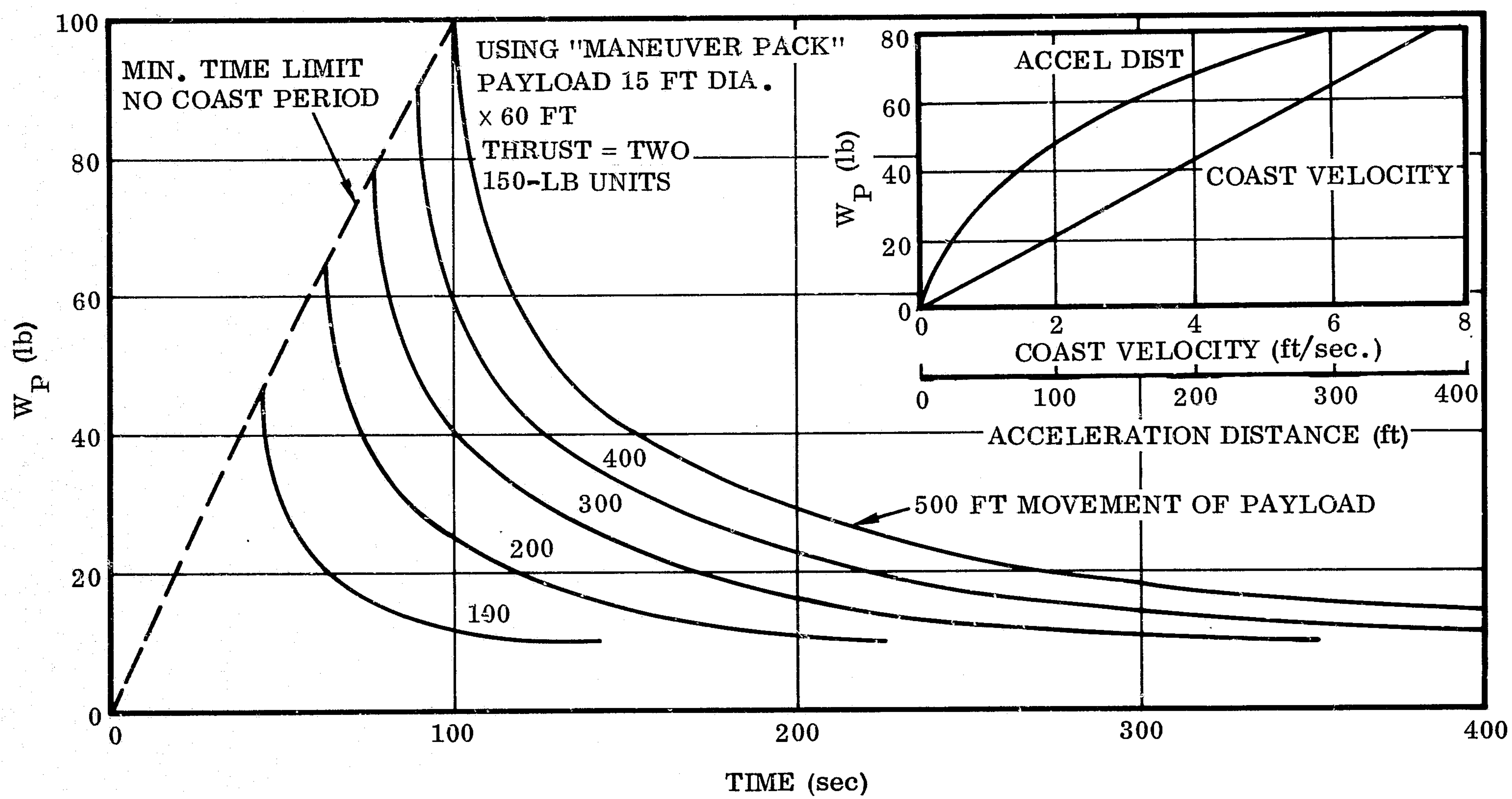


Figure 4-49. Propellant Expended Translating 50, 000-lb Payload

dotted line, is the point where the payload accelerates half way then decelerates for the remaining distance. It can be seen that minimum time operation would be costly from the standpoint of propellant expenditure. The curve in the upper right corner shows how far the payload must accelerate to achieve coast velocity. The same distance is required to decelerate the payload. Coast velocities are shown for the corresponding total propellant expended.

4.5.4 CAPTURE METHODS FOR NONCOOPERATIVE SATELLITE RETRIEVAL. For most missions, rendezvous and docking will be accomplished with a stable target. In some situations, however, the target may be tumbling about an arbitrary axis, due either to the absence or malfunction of attitude control. Such a condition could exist in a satellite retrieval, satellite maintenance, or rescue mission.

Several techniques have been suggested for arresting the angular momentum of such targets. One method would be to provide an independent, redundant attitude control subsystem (MP) on every target, to be activated by the orbiter during rendezvous in the event of a failure in the target primary attitude control system. This would be a safe, simple solution, but does not provide for targets already in orbit.

An approach for existing targets without an MP is to deploy a damping device which would reduce the angular momentum of the target by energy dissipation. Although relatively safe, this approach might require a relatively long time to reduce the target angular velocity to a satisfactory level. In addition, individual solutions may be required for each target type.

Deployment of a device to increase the moment of inertia of the target and thus to reduce angular velocity without affecting angular momentum suffers from disadvantages similar to those of damping devices.

A promising technique involves alignment of the orbiter with the rotational axis of the target, spinning up to the target velocity, grappling the target and then de-spinning. A less risky variation of this method would be to deploy a small tug with a grappling device or net, with an attitude control subsystem controlled from the orbiter.

It is quite likely that more than one technique will be employed for capture of cooperative targets by the orbiter. The optimum method may vary with type of target and target spin rate. In addition, the capture technique may depend on the mission; that is, whether the intent is to retrieve the target, dock with the target, or perform inspection and/or maintenance.

4.6 SHORT-DURATION ORBIT MODULES

The orbiter stage of the space shuttle can serve as a short term orbital laboratory or sensor platform. In this mode of operation the experiment module or sensor equipment would remain in the payload bay as shown in Figure 4-50. When required, sensor viewing can be accomplished by opening the payload doors.

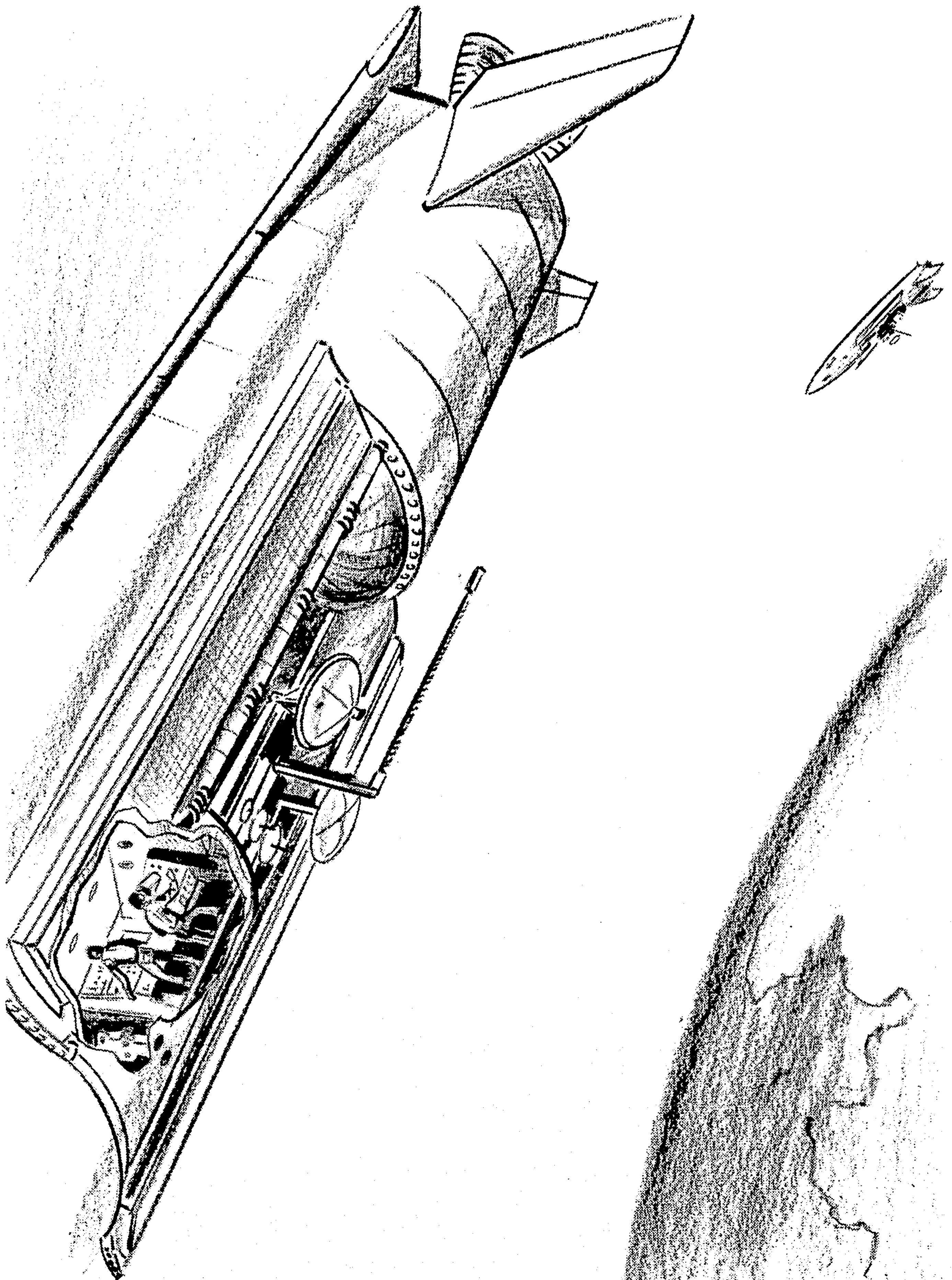


Figure 4-50. Short Duration Orbital Missions

4.6.1 CANDIDATE MODULES. Candidate short duration missions for the space shuttle include:

a. Material science and processing experiments

b. Human factors (onboard centrifuge)

c. Component test and sensor calibration

d. Earth surveys.

Modules for the first three are similar to the experiment modules discussed in Section 4.2.2 except the module would remain in the payload bay during the experiment.

A summary of candidate Earth survey missions, shown in Table 4-12, indicates the nature of the survey and type sensor required. The sensors and required support equipment would also be in modules for installation in the payload bay.

4.6.2 ORBITER REQUIREMENTS. When used as an orbiting laboratory the orbiter will require additional ACS propellant for stabilization and orientation during the operational period. In addition the life support and environmental control system will have to be increased from 7 days to 30 days for both the experiment operators and basic space shuttle crew. This will be provided in the short duration orbit module.

4.7 RESCUE

Since many aspects of a space station design and operation (from the space environment through the crew behavior) are treated on a probability basis, provisions must be made for the occurrence of improbable events. Emergencies on board the space station could require emergency flights of the space shuttle to deliver cargo or evacuate passengers.

The cargo flights will be required to deliver spares and components to repair system failures or damage due to meteoroid impact or collision with other vehicles or space debris. It may also be necessary to deliver expendables such as O_2/N_2 and propellant to make up losses due to the failure. The time available to deliver this cargo cannot be specified due to the multitude of possible emergencies and alternative courses of action available to the station crew. However, if the malfunction or damage is repairable, the crew is probably not in serious danger for at least a few days, due to the compartmentized space station design.

The more critical situations are those requiring the emergency evacuation of crew members or abandonment of the station entirely. Some of the major events which could result in such emergencies are listed in Table 4-13, with possible courses of action. The availability of a rescue vehicle is necessary if the station is to be abandoned immediately.

Table 4-12. Earth Resources Instrumentation and Applications

	Cultural Resources									Natural Resources								Earth Sciences							
	Agricultural: Soils			Agricultural: Crops				Transportation		Fresh Water			Forestry			Wildlife							Natural Disasters		
	Quality	Temperature	Moisture	Quality	Species	Population Distribution	Urban Land Use	Development	Control	Sources	Distribution	Pollution	Distribution	Quality	Marine Life	Distribution	Migration	Cartography	Geological Resources	Geophysics	Geodesy	Oceanography	Floods	Fires	Internal Effects
Metric Cameras	X		X	X	X	X	X	X		X	X		X	X	X	X	X	X	X	X	X	X	X		X
Panoramic Cameras	X		X	X	X	X	X	X		X	X	X	X	X	X	X	X	X	X	X	X	X	X		X
Tracking Telescope	X		X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X		X	X	X	X
Synoptic Cameras						X	X	X		X	X	X	X		X	X		X	X	X		X	X		X
Radar Imager	X						X	X	X	X	X	X	X		X			X	X			X	X	X	
Radar Altimeter/Scatterometer								X		X	X				X				X			X			
Wide-Range Spectral Scanner (O-M)	X	X	X	X	X		X	X		X	X	X	X	X	X	X	X		X	X		X	X	X	X
R Spectrometer					X		X	X		X	X	X			X				X	X		X			X
R Radiometer		X	X				X	X		X	X	X	X	X	X	X	X		X	X		X	X	X	X
Microwave Imager (Passive)			X					X		X	X	X							X	X		X	X		X
Microwave Radiometer		X	X					X		X	X	X							X	X		X	X	X	X
UV Imager/Spectrometer																			X			X			X
Laser Altimeter/Scatterometer																			X	X	X	X	X		
Absorption Spectrometer																			X						
Radio Reflectometer										X	X				X				X			X			
Magnetometer																			X	X	X	X			
Gravity Gradiometer															X				X	X	X	X			
Ground Sensors		X	X						X	X	X						X			X		X		X	X

Table 4-13. Major Space Station Emergencies

Abandon the Station Immediately - Onboard Rescue Vehicle Required

1. Excursion of nuclear power source.
2. Major explosion and/or fire.

Quick Rescue Required (within hours)

1. Severe solar flare.
2. Nuclear burst.
3. Major failure of power subsystem.
4. Major failure of life support subsystem.
5. Sick or injured crew member.
6. Station damaged due to meteroid impact.

Planned Rescue Possible (within days)

1. World situation - threat of war.
2. Major failure of critical subsystem (i.e., station maintenance).

These emergencies impose stringent requirements on the space shuttle if it is to operate as an efficient rescue craft. The first and perhaps the most critical requirement is rapid response. The space shuttle should be capable of being loaded with emergency cargo, where necessary, and launched at the first available window. After the contact is made in orbit, it should be capable of returning immediately.

It may be necessary to transfer personnel from an unpressurized station. This requires either an airlock or the ability to depressurize the space shuttle for the transfer. In addition, the mobility of the personnel will be severely reduced by the inflated space suit. The personnel modules would be used for this rescue mission.

The capability for rendezvous and possible docking with a non-cooperative target may be required when the station is damaged. The only assistance which could be anticipated from the target vehicle would be a simple beacon. Extra-vehicular transfer of personnel and some cargo may be required if the station hub is inoperative.

SECTION 5

SAFETY AND ABORT

Safety (probability of success of intact abort) and probability of mission success goals were established considering safety and cost effects as follows:

Safety	0.999 (1 loss/1000 flights)
Mission	0.97 (30 aborts/1000 flights)

The analysis shows that these goals can be approached or met by fail-operational/fail-safe and fail-safe design approaches and through minimization of fire and explosion hazards.

Safety in flight operations of the FR-3 and FR-4 vehicle concepts is achieved with intact abort. Safety and cost require that the crew, passengers, payload, and vehicle be returned intact following failures requiring abort. The safety and abort analyses show that intact abort is a feasible approach. The basic approach for treating the majority of failures is to provide redundancy to produce a fail-safe system or a fail-operational system. For the FR-3 and FR-4 vehicles, mechanical/electrical subsystems have fail-operational/fail-safe characteristics and the integrated avionics system has fail-operational/fail-operational/fail-safe characteristics. When a failure occurs in a system with fail-operational characteristics there is no abort since the mission can be completed. When a failure occurs in a system at the fail-safe level it is necessary to go to an abort procedure.

The once-around abort procedure reflects action to achieve a high probability of successful intact abort from all failures. For failures during liftoff to staging the booster and orbiter elements will complete the boost phase and stage when the booster propellants are depleted. The vehicles separate and the booster returns to the launch site in a normal manner while the orbiter continues once around the Earth and returns to the launch site.

There are failure situations, however rare, which require immediate abort. Typical of these types of failures are structural failure, thermal protection system failure, and catastrophic situations such as a fire which may require early separation. Following separation, a throttle/burn-dump/reverse flight operation can be used. All remaining propellant is expended through the rocket nozzles and a flight profile is selected to allow the vehicles to return to the launch site.

The FR-3 vehicle, with a 15-3 booster-orbiter engine arrangement, has a relatively low number of mission aborts because it incorporates fail-operational/fail-safe provisions for engines in the booster. The FR-3 can achieve staging with one engine out

because the 7% overthrust capability of the booster engines allows the performance thrust-weight ratio to be maintained. The FR-3 can achieve intact abort with two engines out at liftoff. As the vehicle advances along the boost phase, more engines can be out and intact abort is still possible.

The FR-3 and the FR-4 orbiter engines do not have fail-operational/fail-safe capability during the staging to orbit phase because the weight penalty to provide a 50% overthrust in the three-engine orbiter is prohibitive. The FR-3 and FR-4 orbiters do have fail-operational/fail-safe capability for all on-orbit maneuvers.

The FR-4, with a 9-3-9 booster-orbiter-booster engine arrangement, can achieve staging with one engine out; however, there is a weight penalty because a 13% overthrust is required. This amount of overthrust is outside the presently designed engine propellant utilization control capability. Uprated or added engines are required with associated weight penalties. Because the FR-4 with the 9-3-9 arrangement does not have fail-operational/fail-safe capability for booster engines, mission losses are higher than for the FR-3. The FR-4 has fail-safe provisions for engines and basically the same abort procedures as the FR-3 described above. The intact abort success probability (safety) of the FR-4 is therefore approximately the same as for the FR-3.

Both the FR-3 and FR-4 vehicle concepts incorporate inert gas purging provisions for fuel tank surrounds, rocket engine bay, and payload bay to suppress potential fire or explosion resulting from leakage and subsequent vaporization of fuel (LH_2). Purging with an inert gas is provided during ascent and descent to an O_2 concentration $< 2\%$ by volume for these areas.

Sealed, gas-tight bulkheads separate compartments containing fuels and/or oxidizers and diaphragms seal off hot air and isolate hot surface sources.

5.1 INTRODUCTION

Safety and cost in reusable launch vehicles are the real drivers leading to requirements for a high probability of successful abort. Crew, passengers, payload, and the vehicles must be returned intact to make the reusable launch vehicle concept economically attractive. A safety and cost analysis conducted for the space shuttle (Section 5.2) was accomplished on this premise; i.e., intact abort. This analysis established safety and mission success goals which were used as a guide for the study. Basic questions which must now be addressed are:

What makes the space shuttle unsafe?

What action must be taken to change an unsafe situation into a routine abort operation?

How is safety improved?

What are the interfaces of safety with weight, operations, and mission success?

These questions were answered by conducting a gross failure and mission termination analysis (Section 5. 3) with consideration given to:

a. Probability of occurrence of failures of subsystems, propulsion systems and structure during the mission.

b. Abort options following these failures.

c. Availability of landing sites for aborted flights for several launch azimuths.

d. The fire and explosion hazard potential of the stored propellants.

e. Intact abort, redundancy, and escape.

This analysis leads to:

a. Definition of abort procedures from the various flight trajectory phases (Section 5.4),

b. Design requirements for the vehicle (Section 5.5), and

c. Design requirements for the minimization of the fire and explosion hazard (Section 5.6).

The effect of engines on safety and mission success was studied using a range of assumed engine reliabilities and the application of fail-operational and fail-safe criteria to the booster and orbiter engines (Section 5. 7).

Finally, the safety, abort, and mission success characteristics of the FR-3 and FR-4 vehicles are defined (Section 5. 8).

5.2 SAFETY, MISSION SUCCESS, AND COST INTERFACE

The following analysis and philosophy were used to establish goals for the initial FR-1 safety study. The information presented in this section also applies to the FR-3, and FR-4 concepts.

Safety for space shuttle concepts introduces a new factor, not found in expendable systems, of determining the probability of successful recovery of the vehicles since the vehicles are a priori designed for reuse. Expendable systems are evaluated basically in terms of probability of mission success and safety; reusable systems must consider in addition the recovery of vehicles following abort.

A vehicle that can execute an abort and intact return represents a safe vehicle for crew and passengers. Further, operational costs are minimized if vehicles are returned intact and are available for reuse. The relationship between safety and cost is shown in Figure 5-1, which is a plot of the cost of vehicle losses, the cost of payload losses, and the cost to relaunch (incompleted missions) versus losses per 1000 flights. These data were based on preliminary cost data for a smaller payload FR-1 and should be viewed qualitatively.

As shown, operation costs are reduced as safety is increased. Vehicle losses and lost payloads are a major cost factor while the cost of relaunch (incomplete missions) is a small cost consideration. Figure 5-1 was developed on the basis of 1000 flights using lower cost vehicles and higher cost (2% of vehicle cost) relaunch to emphasize the point. Both safety and cost tend to drive the vehicle design toward lower losses, approaching zero as a limit. However, this is not the complete story because, as shown in Figure 5-2, there is a minimum cost vehicle design which tends to bucket at losses slightly less than 1/1000 due to the added cost of RDT&E to achieve lower losses (higher reliability). Figure 5-2 shows the cost of vehicle losses, and RDT&E added cost.

A goal in the range of 1 to 2 losses per 1000 flight (0.998 to 0.999) is indicated by Figure 5-2 on the basis of costs. However, when considering the lives involved, the design objective should be for lower losses per 1000 flights. A goal of 1/1000 is used for discussion purposes to show safety and cost sensitivities.

This analysis established the goals given in Table 5-1. They include failures due to mechanical and electrical malfunctions as well as the hazard of fire and explosion:

Table 5-1. Reliability Goals

Safety	> 0.999 (1 losses/1000 flights)
Intact Abort Success	> 0.999 (1 losses/1000 flights)
Mission Success	> 0.97 (30 aborts/1000 flights)

The safety/cost interface in reusable launch vehicles can be summarized:

a. The cost of lost vehicles and lost payloads is a major operational cost factor.

b. The cost of incompleted missions (aborted mission) is a minor operational cost factor for vehicles having a high probability of successful intact abort.

c. To be economically feasible, reusable launch vehicle concepts must achieve a probability of successful intact abort $\geq$ 0.999 (losses < 1/1000).

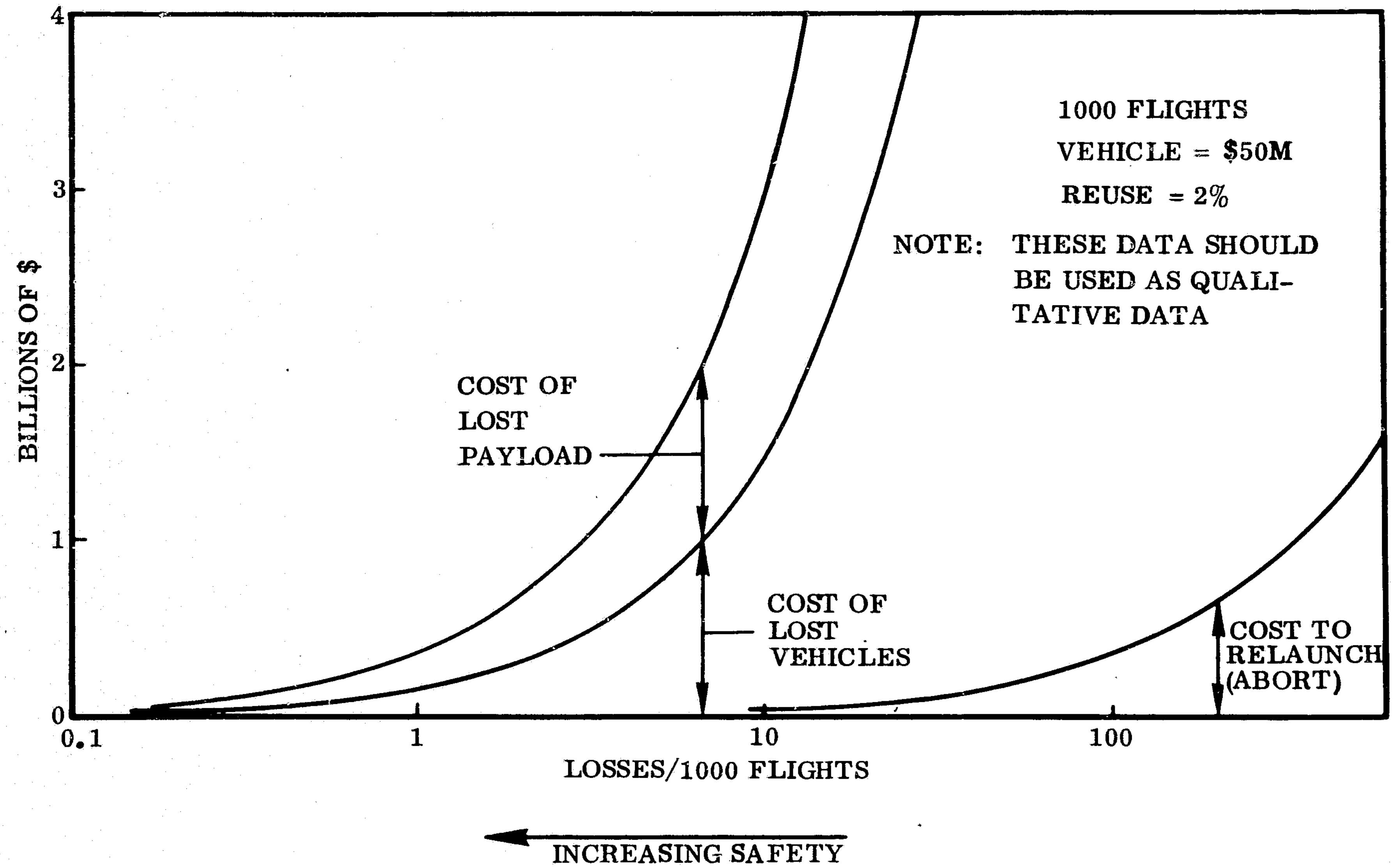


Figure 5-1. Cost of Lost Vehicles

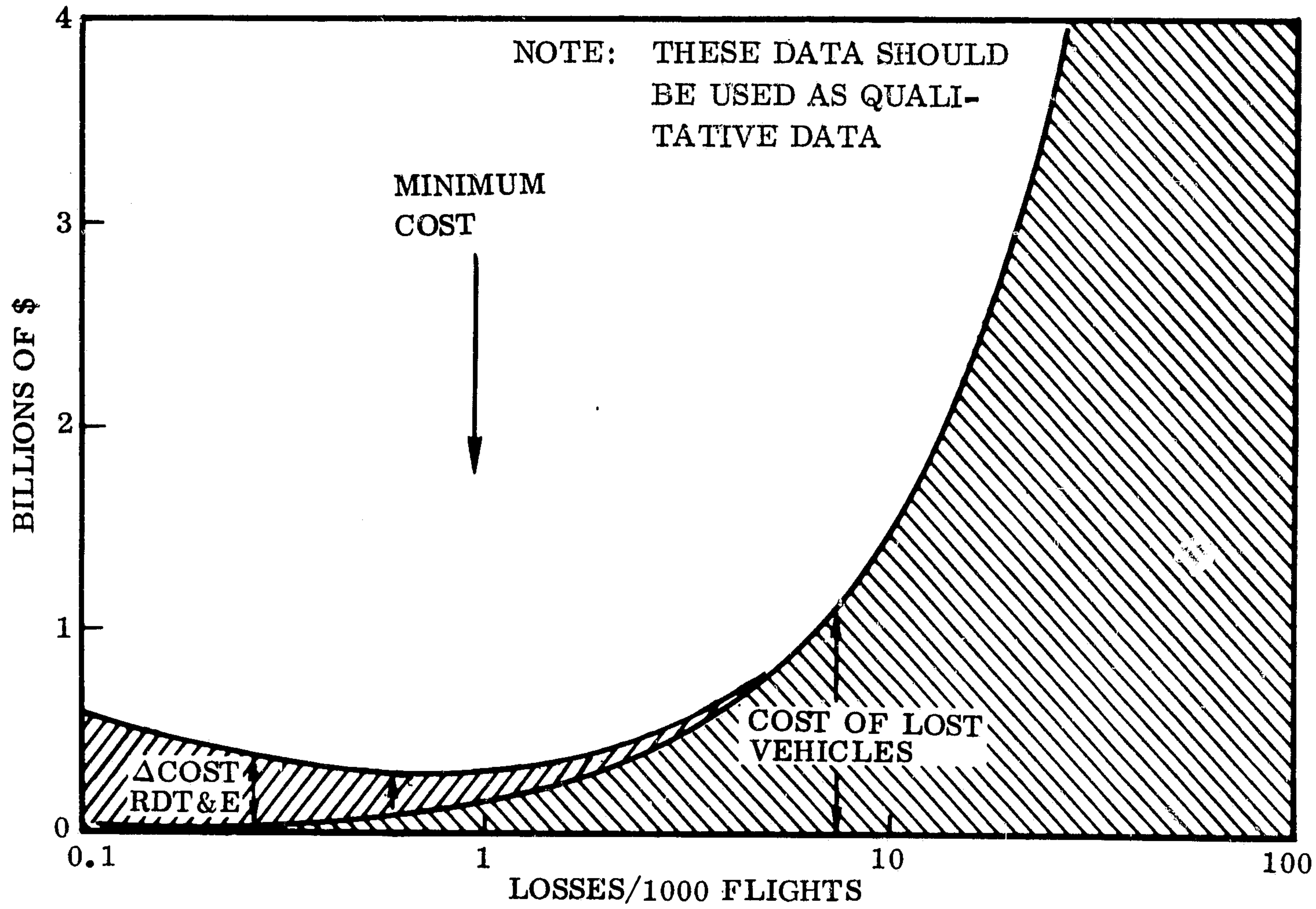


Figure 5-2. Minimum Cost System

d. A vehicle designed with losses <1/1000 has the potential for improved safety at an order of magnitude cost reduction compared with vehicles designed for 20/1000 losses, which is normally an acceptable mission goal for an expendable system.

One of the desired vehicle characteristics for this study was intact abort. This is a reasonable requirement for the following reasons:

a. Safety. Safe intact return of the vehicles represents safety for crew and passengers. With a high probability of success of vehicle recovery, safety objectives can be achieved without the use of escape systems.

b. Cost. Operations cost and safety are major drivers leading to the intact abort concept. To make the fully reusable concept economically feasible, vehicle return success must approach $\geq$ 0.999 (< 1/1000 losses), which provides a base for evaluation of improved safety.

c. Reusability. Unlike expendable launch vehicles the reusable vehicles have reuse capability which inherently provides intact abort potential.

5.3 FAILURE AND MISSION TERMINATION ANALYSIS

A gross failure and mission termination analysis and a hazards analysis were conducted on an early FR-1 configuration. The basic objective was to establish basic abort philosophies, mission termination procedures and subsystem design requirements including redundancy. The data and information developed is qualitatively but not quantitatively applicable to the FR-3 and FR-4. The basic once around abort and immediate abort concepts were developed from this analysis using previously established mission success goals and safety goals.

The data and information developed were used to guide the vehicle and vehicle systems design tasks including an investigation of propellant dumping.

The approach used in the gross failure and mission termination analysis for mechanical failures in subsystems was:

a. Establish failure rates from historical data and estimates.

b. Apply weighting factors to account for differences in mission phase stresses.

c. Determine points in the mission when failures are most likely to occur (distribute failures into the mission phases).

d. Determine the consequences of major subsystem failures and investigate abort procedures which lead to successful intact recovery of vehicles.

e. Make design improvements and develop operational abort procedures.

5.3.1 HISTORICAL FAILURE DATA AND ENVIRONMENTAL WEIGHTING. Failure data for the failure analysis were developed using failure rates based on mean-time-between failure (MTBF) values obtained from:

a. Historical data on, for example, engines (F-1, J-2, RL-10, H-1).

b. Previous study reports such as: ALSS, Saturn V Reliability Analysis Model, SA-501 (Marshall — Sept. 7, 1965), Atlas-Centaur AC-13 Reliability Assessment Report.

c. Estimates made for unique components and systems such as wing deployment, turbojet engine deployment, and the separation system.

Operating time (t) for each subsystem is based on required operating time to perform the specified mission. Figure 5-3 shows values used in the analysis.

Environmental weighting factors were applied to account for the effect of different environmental stresses on failure rate. Weighting factors (k) were applied for each mission phase as shown in Table 5-2.

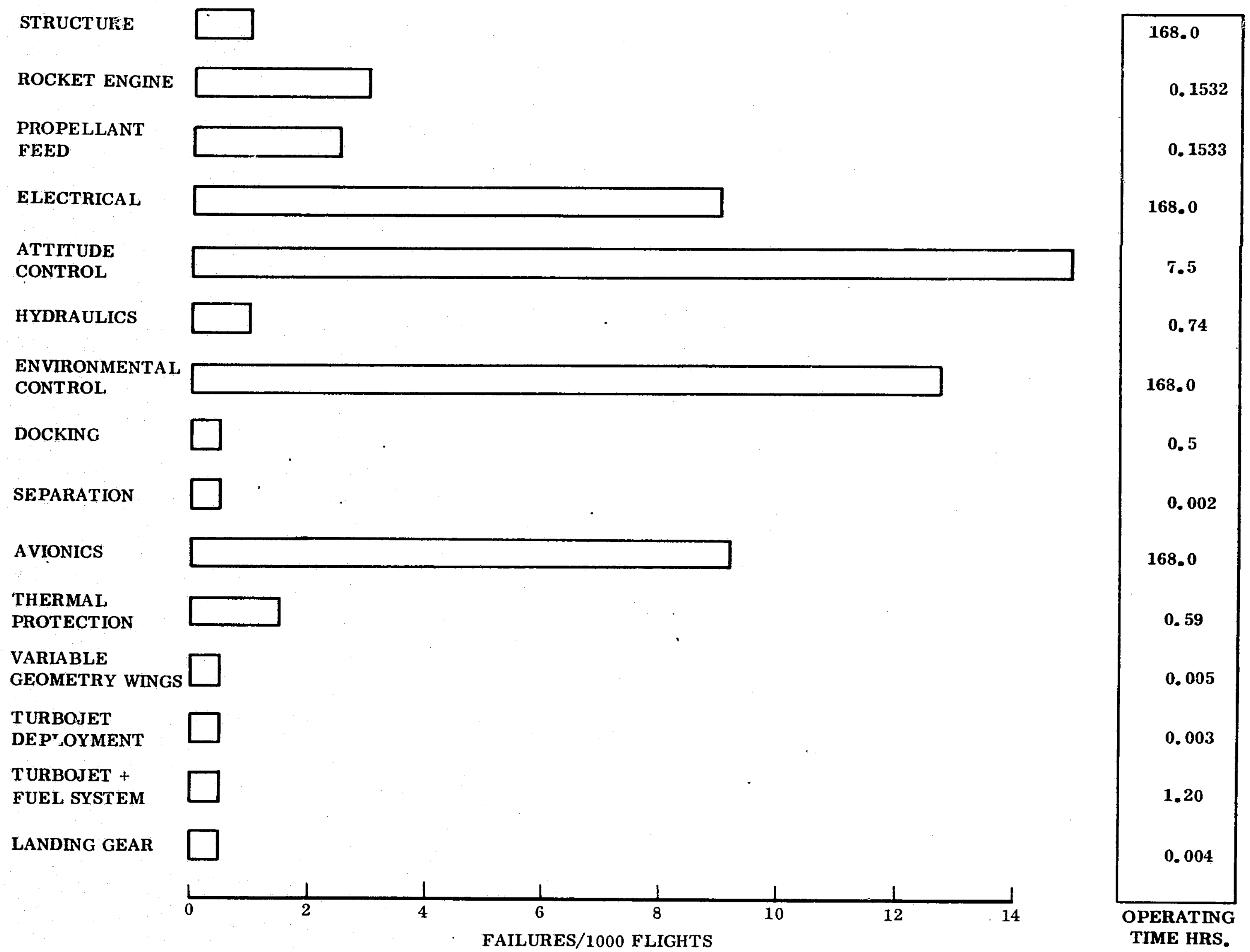


Figure 5-3. Historical Failure Data

Table 5-2. Environmental Weighting for Mission

Trajectory Phase	Time = t (hr)	Weighting Factor = k	kt
Boost to Staging	0.05	500	25.0
Staging to Orbit	0.061	500	30.5
Orbit (Coast)	24.04	1	24.0
Orbit (Rocket)	0.041	100	4.2
Orbital Missions	144.0	1	144.0
Docking	0.50	1	0.5
Entry from Orbit	0.59	100	59.0
Flyback and Landing	1.17	1	1.2

The expression $kt/\Sigma t$ was used to distribute failures throughout the different mission phases.

5.3.2 MISSION FAILURE DISTRIBUTION. Failures were distributed into the mission phases as shown in Table 5-2.

Figure 5-4 shows the total expected failures distributed among the mission phases. Failures are for the baseline vehicle subsystems before evaluation of the consequences of the failures and before incorporation of recommended operational and design improvements and therefore do not represent vehicle losses. Failures in each mission phase were calculated using the following basic expression:

$$\text{Failure} = \sum_{i,\, j=1}^{n} (n_j) \left(\frac{1}{MTBF_j}\right) (\Sigma t_i) \left(\frac{k_i t_k}{\Sigma k_i t_i}\right)$$

where

n_j = jth component or subsystem operating in series (i.e., engines)

$MTBF_j$ = Mean-time-between failures of the jth component or subsystem in hours/failure (from historical data)

t_i = operating time (hours) in ith mission phase

k_i = environmental weighting factor for the ith mission phase (accounts for the stress level of a given trajectory phase)

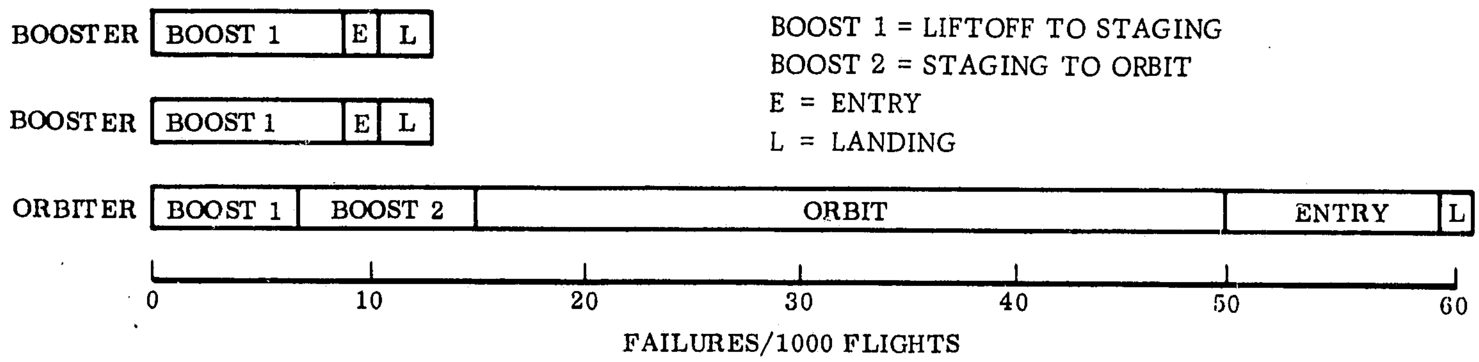


Figure 5-4. Failures In Subsystems During Mission Phases

Failures in subsystems of about 84 per 1000 are predicted. Vehicle losses are substantially less. Abort and safe recovery are therefore of primary concern.

5.3.2.1 Boost-to-Staging Failures. Failure distribution is shown in Figure 5-5 for the first boost phase (liftoff to staging). Shown here are total expected failures per 1000 flights before evaluating the consequences of the failure and before incorporation of recommended operational and design improvements.

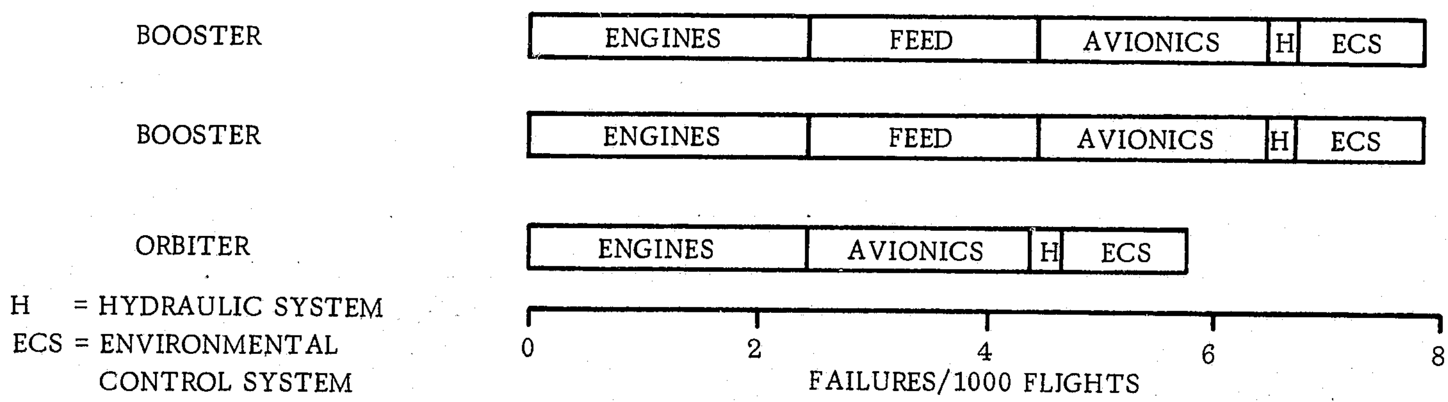


Figure 5-5. Boost-to-Staging Failures in Subsystems

5.3.2.2 Staging-to-Orbit Failures. Figure 5-6 shows total expected failures before evaluation of the consequences of the failures and before incorporation of recommended improvements. The first stage boost cycle and stage separation have been completed.

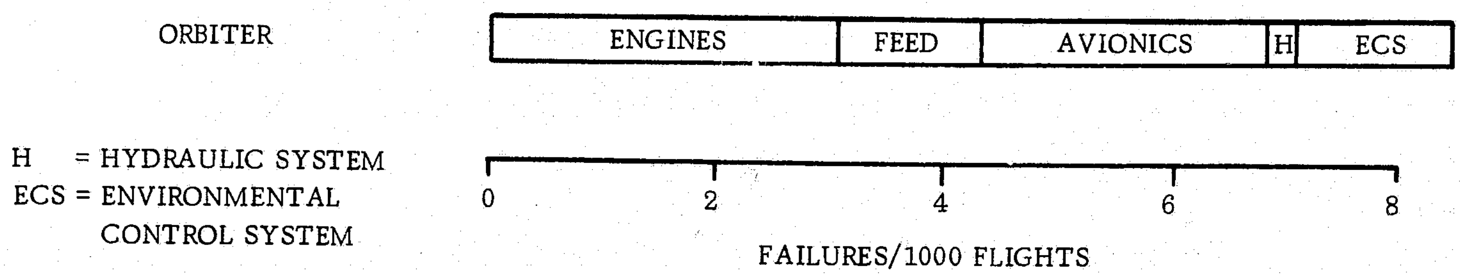


Figure 5-6. Staging-to-Orbit Failures in Subsystems

5.3.2.3 In-Orbit Failures. Figure 5-7 shows expected failures for the orbit phase.

In the figure:

E	= engine
F	= feed system (propellant),
ACS	= attitude control system for orbiter
ECS	= environmental control system
AV	= avionics
D	= docking

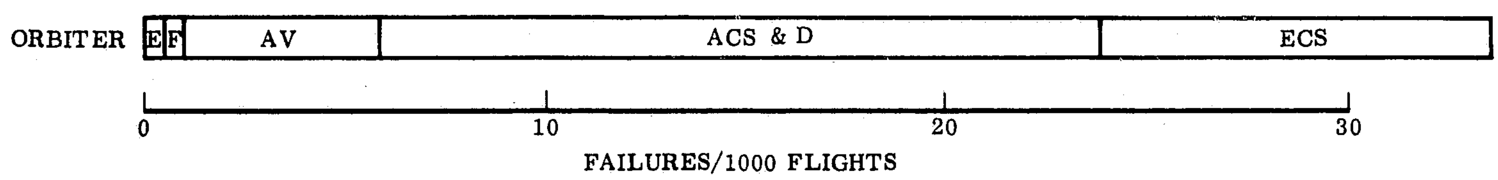


Figure 5-7. Failures in Orbit in Subsystems

5.3.2.4 Entry Failures. Figure 5-8 shows total expected failures before evaluation of the consequences of the failure and before incorporation of recommended improvement.

In the figure:

T/S	= thermal protection system and structure
E	= environmental control system
S	= structure
H	= hydraulic system
AV	= avionics and electronics

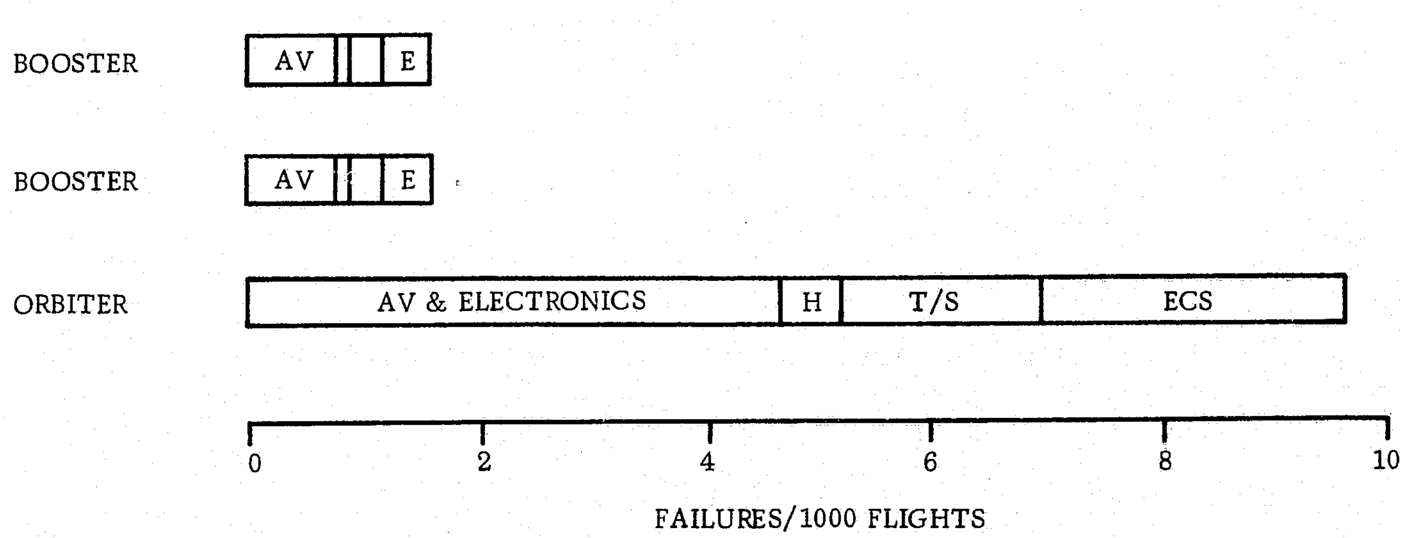


Figure 5-8. Entry Failures in Subsystems

5.3.2.5 Flyback and Landing Failures. Figure 5-9 shows total expected failure before evaluation and improvement.

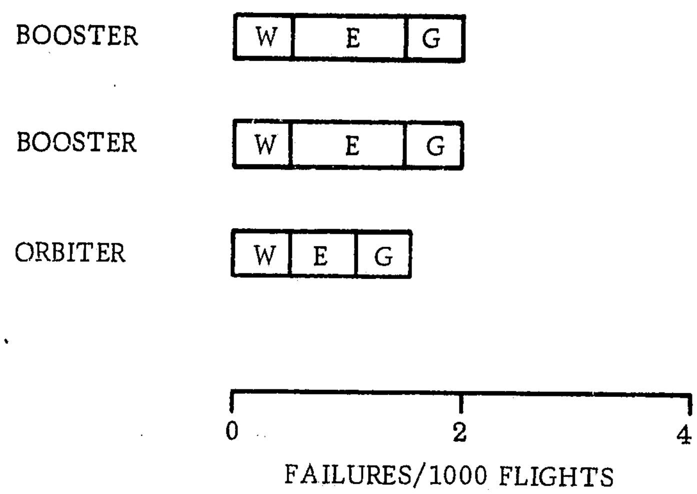


Figure 5-9. Flyback and Landing Failures in Subsystems

In the figure:

W = wing deployment

E = turbojet engine deployment and operation

G = landing gear system

5.3.3 CONSEQUENCES OF FAILURES AND ABORT PHILOSOPHY. Each mission phase was examined and the failures were evaluated to determine the effect on the vehicle and what corrective action in the way of changes in operation or design changes (e.g., adding an engine) could be taken if the consequences of the failure meant an aborted mission or loss of life. Emphasis in the failure analysis was on crew and vehicle recovery (crew recovery from a safety standpoint and vehicle recovery from an economic standpoint).

Systems were examined to assure that no single failure resulted in loss of life. This was done using the airplane systems design approach wherein backup systems are used to accomplish safe return. Investigations were also made to eliminate or reduce the number of time critical failures.

The abort philosophy used in the failure analysis is summarized below. Priority for crew and passenger safety was the first consideration.

a. First priority: Save crew and passengers.

b. Second priority: Save payloads.

c. Landing site priority:

   1. Return to launch site.

   2. Once around-return to launch site.

   3. CONUS landing sites.

   4. Available landing sites.

   5. Survival/rescue.

d. Prior to docking: Return by earliest (low stress) route.

e. After docking: Continue to landing.

5.3.4 ABORT GLIDE FOOTPRINTS. One of the consequences of failure is availability of landing sites.

Figure 5-10 shows the abort/glide footprints, when launching from ETR, for launch azimuths 0 to 90 degrees. A hypersonic lift drag ratio of 1.9 is assumed. Example footprints are shown for abort velocities of 6,000, 10,000 and 15,000 fps for a direct injection into a 55-degree inclination orbit, with a launch azimuth about 37 degrees. For this orbit, landing sites are available along the Eastern sea board; however, more easterly launches are entirely over water. Landing site availability is therefore strongly dependent on launch azimuth and abort velocity.

Figures 5-11, 5-12, and 5-13 present landing site availability for three launch azimuths from ETR.

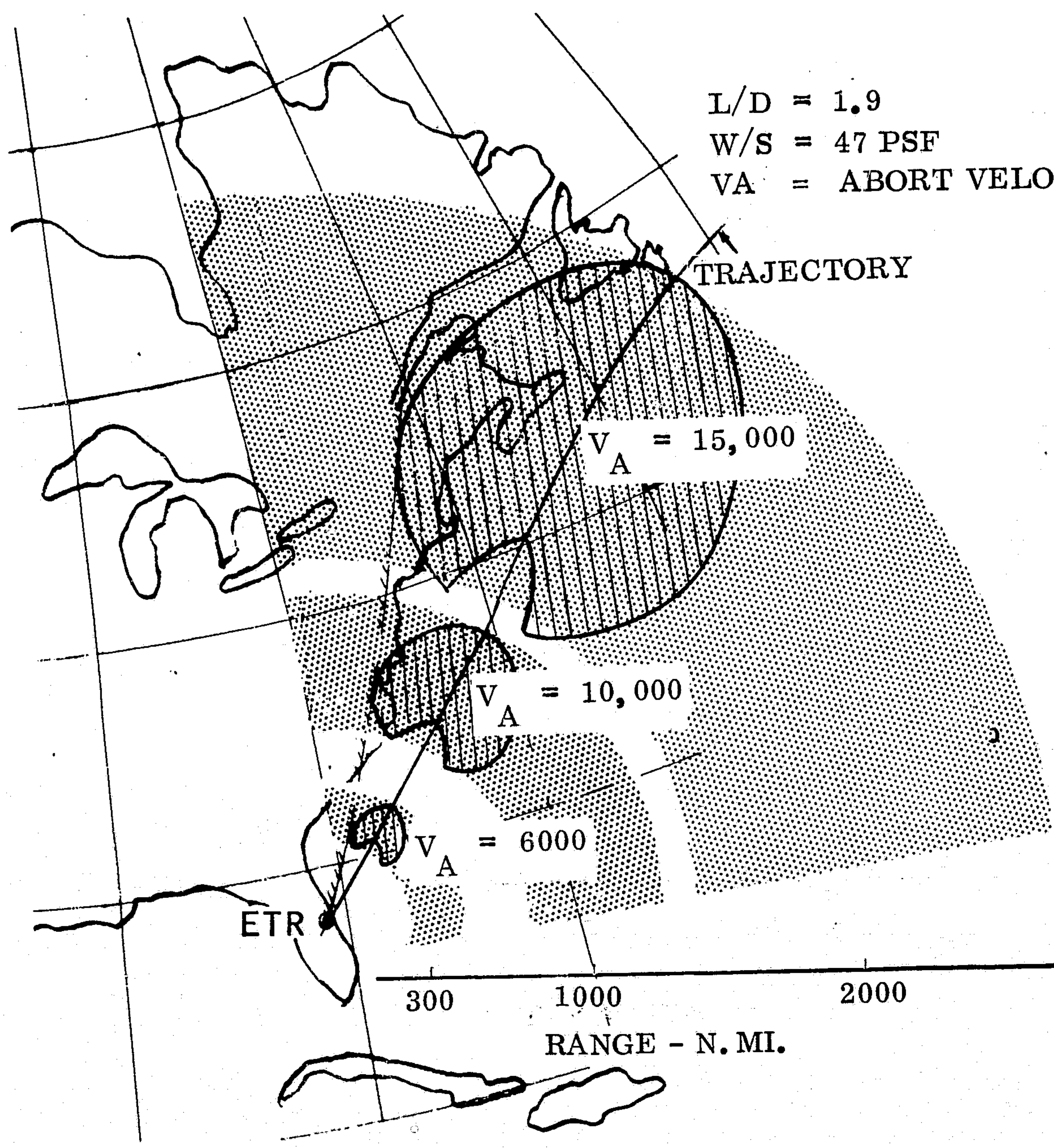


Figure 5-10. Abort/Glide Footprints

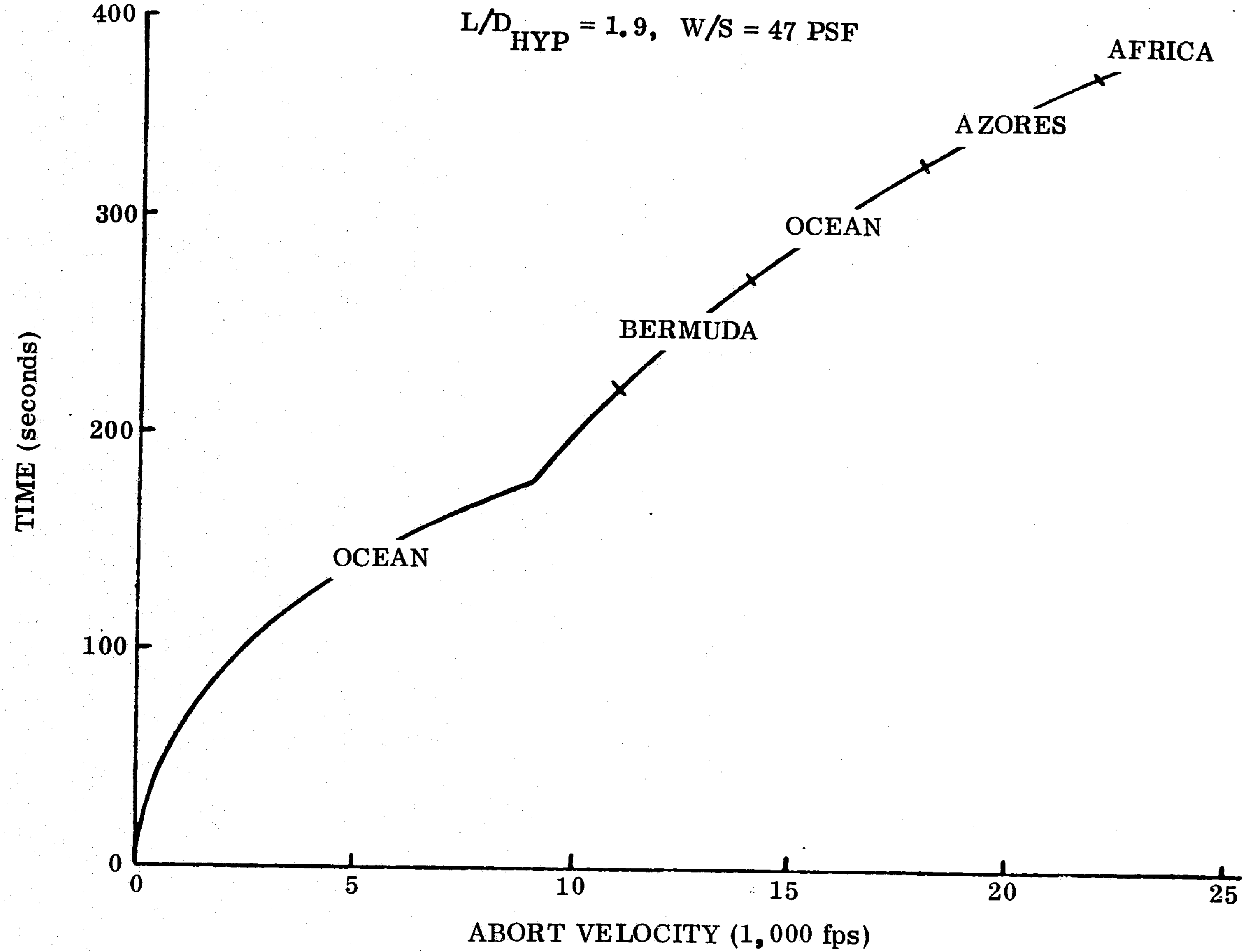


Figure 5-11. Landing Site Availability (Glide) 30° Orbit From ETR

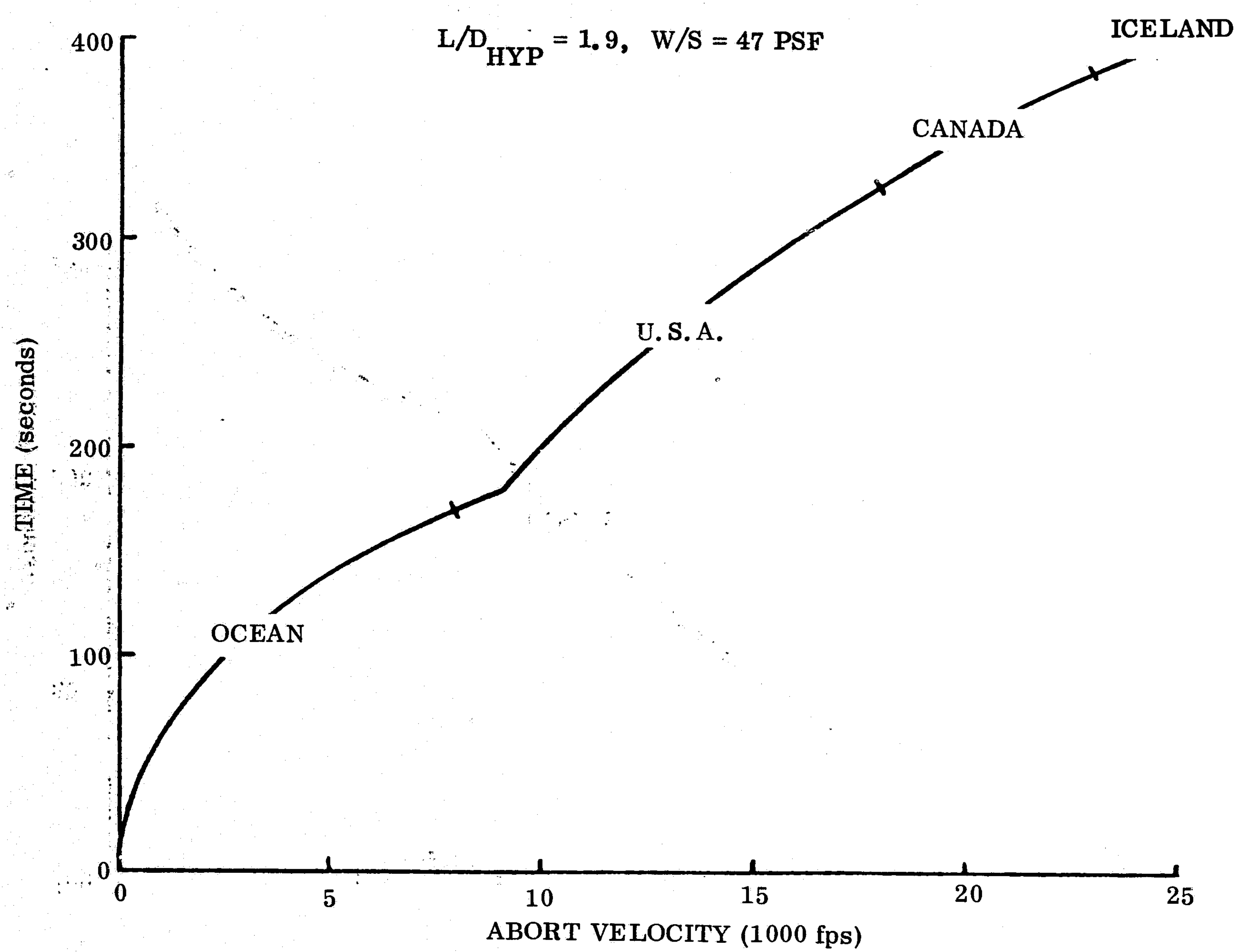


Figure 5-12. Landing Site Availability (Glide) 55° Orbit From ETR

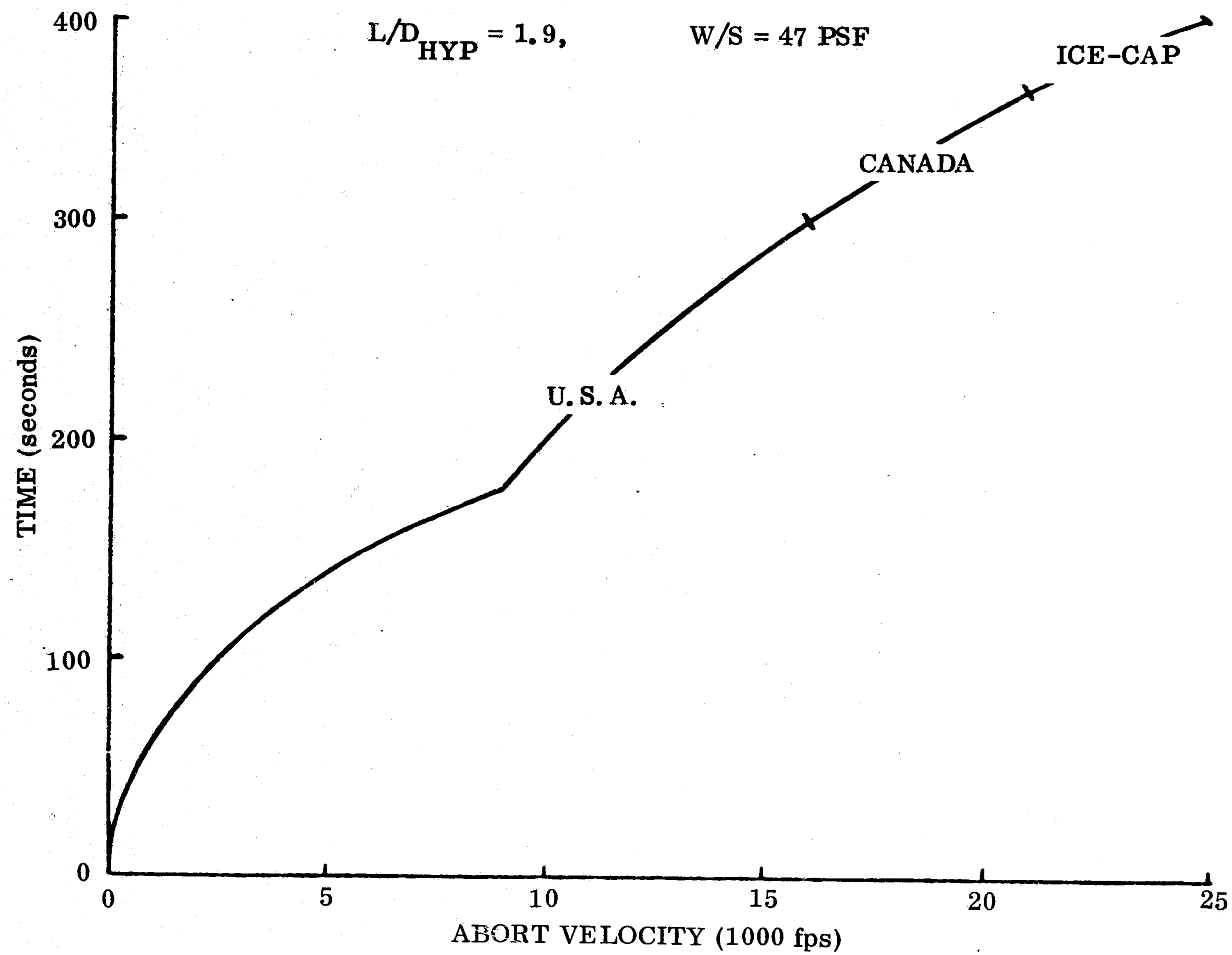


Figure 5-13. Landing Site Availability (Glide) 90° Orbit From ETR

5.3.5 FAILURES. The situation, effect, corrective action, and results were investigated for each major failure that has a bearing on safety and vehicle recovery. This information is presented on the following pages.

5.3.5.1 Loss of Booster Engine. Failure of a boost engine results in reduction of thrust to weight (T/W) from 1.32 to 1.13, and in a requirement for a 5 degree gimbal angle trim capability to hold thrust vector through vehicle c.g. With this T/W, vehicle cluster can accelerate to some suitable staging velocity and attitude; however, structural failure will occur (increased angle of attack caused by 5 degree gimbal angle) if the engine is lost before maximum wind shear is encountered, and some propellant will be trapped in the booster with the failed engine. The probability of occurrence (P_O) of loss of one booster engine is estimated at about 5/1000 flights (a significant number), which for the baseline design would result in loss of crews and vehicles.

Recommended corrective action is:

a. Provide propellant crossfeed capability, which will deplete all booster tanks equally.

b. Provide structural beefup to withstand increased loads during travel through the maximum wind-shear loading condition.

The results expected with these recommendations incorporated are: (1) the five fatality failures become aborts (no fatalities), (2) orbiter will go once around Earth and land at the launch site, (3) boosters will continue to burn off propellant uniformity, separate, and return to the launch site, (4) added structural beefup will allow cluster to pass through maximum αq without failure, and (5) all crews and vehicles are recovered.

Loss of two boost engines (0.02/1000) is considered negligible. The following alternate solution was investigated, but is not recommended:

a. Provide propellant dump capability in boosters and orbiter.

b. Deplete propellant in both boosters at equal rate by burning and dumping (booster with failed engine).

c. Continue flight to some staging velocity and separate with boosters empty.

d. Boosters return to launch site.

e. Orbiter executes "reverse" maneuver as follows: with staging velocity > 500 ft/sec, reduce this velocity to zero and return to launch site. After flying downrange to some point and approaching a flight path angle (γ) of 90 degrees, reverse direction and head for launch site. Arrive over launch site at subsonic velocity and with zero propellants.

5.3.5.2 Loss of An Orbiter Engine. Following the failure of an orbiter engine during boost phase, the vehicle cluster can accelerate ($T/W = 1.13$) to some staging velocity and separate. The engine gimbal angle required to correct moment is <5 deg. Booster propellants must be depleted equally before separation is attempted. This can be accomplished with the use of a crossfeed propellant system. With booster propellant depleted at staging, the boosters can return to the launch site. Orbiter now has sufficient ΔV to go once around the Earth.

The probability of occurrence of failure for this case is estimated to be 2.5/1000 flights, which would result in loss of all crews and vehicles for the baseline configurations.

Recommended corrective action is: provide propellant crossfeed to enable depletion of booster tanks equally. With this recommendation incorporated:

a. The 2.5 fatality failures become aborts (no fatalities).

b. Booster will burn off propellant equally, separate, and return to the launch site.

c. Orbiter will separate at some velocity, go once around, and land at the launch site.

d. All crews and vehicles are recovered.

An alternate solution was investigated, but is not recommended:

a. For boosters (same as above recommendation).

b. For orbiter, execute "reverse" maneuver as described previously.

5.3.5.3 Loss of Propellant Feed. The loss of propellant feed from one booster during liftoff-to-staging results in the loss of three engines. The result of this failure is catastrophic and is time critical as shown because T/W is reduced to 0.66 and the gimbal angle required to correct the offset moment is 14 deg. Further, propellant is trapped in the failed booster. The probability of occurrence of this mode of failure is estimated to be 4.5/1000 flights. Recommended corrective action is: provide a backup system which will activate within one second to provide propellant feed to all three engines.

With this recommendation incorporated, the 4.5 fatal failures become 4.4. aborts with approximately 0.1/1000 flight fatalities remaining.

Because of the critical nature of this failure and the uncertainty of the practicality of a backup system, alternate approaches were investigated. They are not recommended, however.

Case I. Failure occurs just off pad and before maximum αq.

Recovery procedure:

a. Switch to orbiter propellant feed in 1 second and throttle all engines.

b. Dump failed booster propellant at + "g".

c. Burn and dump good booster propellants.

d. Burn and dump orbiter propellants.

e. Separate.

f. Boosters flyback to launch site.

g. Orbiter flyback to launch site (must add turbojet + flyback fuel).

Vehicle modifications required:

a. Provide crossfeed from orbiter to left or right-hand cluster of three eniginés.

b. Provide propellant dump provisions for boosters and orbiter.

c. Provide flyback turbojet + approximately 250 n.mi. flyback fuel for orbiter.

With these procedures and modifications, 4.5 fatalities per 1000 flights are reduced to about 0.1 per 1000 flights and the remaining 4.4 become aborts with all crews and vehicles recovered by landing at the launch site.

Case II. Failures occurring in the maximum αq area.

Recovery procedure: Same as above, except orbiter now has sufficient ΔV in remaining propellant to land downrange.

Vehicle modification required: Same as Case I except turbojet and fuel not required.

Case III. Failures occurring after maximum αq.

Recovery procedure: Same as Case II.

Vehicle modification required: Same as Case II.

Case IV. Failures occurring near normal staging velocity.

Recovery procedure: Same as Case I.

Vehicle modification required: Same as Case II.

5.3.5.4 Loss of Engine Gimballing. Loss of engine gimballing is time critical and failures are catastrophic. Estimated probability of occurrence is 2.7/1000 flights.

Recommended corrective action is:

a. Provide backup systems in guidance, hydraulics, and electrical power and distribution with one second reaction time.

b. Provide onboard checkout system to detect that failure has occurred.

c. Continue flight through staging with boosters returning to the launch site and orbiter going once around and landing at launch site.

With these recommendations incorporated:

a. The 2.7 catastrophic failures become mission aborts and the probability of a catastrophic failure occurring is approximately 0.01/1000.

b. Crews and vehicles return to launch site with orbiter going once around the Earth.

5.3.5.5 Loss of Environmental Control. The approach here is that of airplane design philosophy. Backup systems are used to provide safe return in the event of failure of the primary system. The probability of occurrence of environmental control system failures is estimated to be about 4.0/1000 flights.

Recommended corrective action is:

a. For booster, provide 100% backup system.

b. For orbiter, provide backup for a once around flight and land at the launch site.

c. Provide onboard checkout and failure detection system allow detection of failure.

With these recommendations incorporated:

a. Boosters will complete boost phase and fatalities are reduced to approximately 0.01/1000 flights. Boosters will land at launch site.

b. Orbiter will complete boost phase, continue once around, and land at the launch site.

5.3.5.6 Separation. The situation and the effect resulting from failure to separate is catastrophic. The failure rate is estimated at 1.5/1000 flights, a significant number.

Recommended corrective action is: provide independent backup separation modes for mechanical, electrical, and pyrotechnic systems.

With this recommendation incorporated the mission may be completed and catastrophies are eliminated.

5.3.5.7 Loss of an Orbiter Engine. Loss of one orbiter engine in the staging-to-orbit boost phase results in loss of roll control, reduced thrust, and a gimbal angle of 5 deg being required to maintain the thrust vector through the vehicle c.g. Frequency of failures is estimated to be 3.2/1000 flights. These are classed as fatalities when no landing site is available.

Recommended abort procedure is:

a. Use remaining propellant to go once around and land at the launch site.

b. Provide dump capability - dump unused propellants.

c. Use ACS for roll control.

5.3.5.8 Loss of Propellant Feed. The loss of propellant feed in the orbiter during the staging-to-orbit boost phase results in loss of thrust from both engines. The effect of loss of propellant feed is total loss of thrust. The problem is the propellant remaining that must be dumped before landing. Propellant dumping is not possible under the resulting zero-g condition. Frequency of failure is estimated at 1.4/1000 flights, and failures are classed as catastrophic (loss of crew and vehicle).

Recommended corrective action is to provide a backup feed system. (Again, no single failure should result in loss of propellant flow to both engines.) Provide onboard check-out to detect failure and permit maintenance action before next flight (airplane design and operation philosophy).

With recommended corrective action the 1.4 catastrophic failures become aborts and the vehicle can return to the launch site by going once around the Earth.

5.3.5.9 Loss of Gimbal. Estimated probability of occurrence of failure is 1.0/1000 flights and it is catastrophic. The failure is time critical.

Recommended corrective action is:

a. Provide a backup capability in hydraulics, guidance, and electrical power and distribution capable of sensing and activating in 5 seconds.

b. Provide means of dumping orbit maneuver propellants.

c. Provide onboard checkout and sensing system to detect failure.

With these recommendations incorporated:

a. The 1.0/1000 catastrophies become aborts.

b. Mission is aborted and a once-around the Earth maneuver is executed with orbiter returning to launch site.

5.3.5.10 Loss of an Orbiter Engine. Thrust is reduced to 1/2. However, the remaining engine can develop the required impulse for orbital maneuvers to enable return from orbit. Probability of occurrence of failure is 0.5/1000 flights and the failures are classed as aborts.

No corrective action is required. If failure occurs before docking, wait in orbit for suitable entry window and land at launch site. If failure occurs after docking, continue entry.

5.3.5.11 Loss of Attitude Control. The vehicle cannot be properly oriented resulting in a catastrophic condition. Failure rate is estimated at 18.1/1000 flights, and they are classed as fatal failures.

Recommended corrective action for this failure is to provide standby systems that are capable of orienting the vehicle (at reduced pitch, yaw, and roll rates) to allow retro-firing.

With the recommendations incorporated catastrophic failures are reduced to less than 0.1/1000 flights. Procedure following failure is:

a. Abort mission if failure occurs before docking. Wait in orbit and land at launch site.

b. Continue entry if failure occurs after docking.

5.3.5.12 Loss of Thermal Protection System. The effect is loss of structure, which is fatal. Failure rate is estimated at 0.5/1000 flights for the boosters and 1.5/1000 flights for the orbiters.

Recommended corrective action:

a. Design fail-safe attachment or under-layer of ablation.

b. Provide means of detecting deficiency before next flight.

5.3.5.13 Loss of Aerodynamic Control. Loss of aerodynamic control during entry is catastrophic for all phases of flight. The greatest number of failures (estimated at 2.9/1000) will occur during entry.

Recommended corrective action is: provide backup systems for hydraulics, guidance, and electrical power and distribution. Provisions to detect failure are required so that maintenance action can be taken before the next flight. Automatic switching to standby mode is required because of the time critical nature of this failure. The 2.9 fatal failures are then eliminated.

5.3.5.14 Subsonic Wings Fail to Deploy. The effect of this failure is catastrophic. Failure rate is estimated at 0.5/1000 flights for each vehicle, making a total of 1.5/1000 flights for the boosters and one orbiter.

Recommended corrective action is to provide fail-safe backup modes in the electro/mechanical/hydraulic system.

5.3.5.15 Loss of Turbojet Thrust. The effect of loss of 50% of turbojet thrust in the baseline vehicle results in thrust < drag which means range is severely reduced. Such failures are catastrophic for booster for launch azimuths > 15 deg. Failures are summarized below and consider deployment and loss of thrust during engine operation.

Vehicle Failure/ 1000 Flight	Failure to Deploy	Failure During Operation	Total
Boosters (2)	1.00	0.70	1.70 Fatal
Orbiter (1)	0.05	0.05	0.10 Fatal

These are all classed as fatal failures. The value of 0.05 deploy failures for the orbiter is based on an estimated go-around requirement of once in 10 landings.

Recommended corrective design action:

a. Provide redundant electro-mechanical, structural, pivot, and starting systems.

b. Provide flyback capability with one engine malfunctioned.

5.3.5.16 Landing Gear. The expected failures of the landing gear deployment function are calculated to be 1.5/1000 flights for the three vehicles (2 boosters and 1 orbiter) returning. One approach to correction is to provide a backup energy source in which case failure probability is reduced to a negligible number. An alternate approach is to provide a keel structure capable of providing for a successful belly landing. In this case, purging of propellant tanks is required.

5.3.6 DESIGN IMPROVEMENTS. Figure 5-14 shows the effect of incorporation of recommended operational and design improvements discussed previously. Incorporation of improvements results in catastrophic failures becoming aborts or a significant reduction in total number of catastrophies.

For example, the fourth item listed is propellant feed. Total failures which are catastrophic are shown equal to 5.9. With the recommended backup system incorporated, the 5.9 catastrophies become 5.8 aborts and 0.1 catastrophies.

As a second example, aero surface controls, the seventh item listed, with improvements has a probability of occurrence of failure which is negligible.

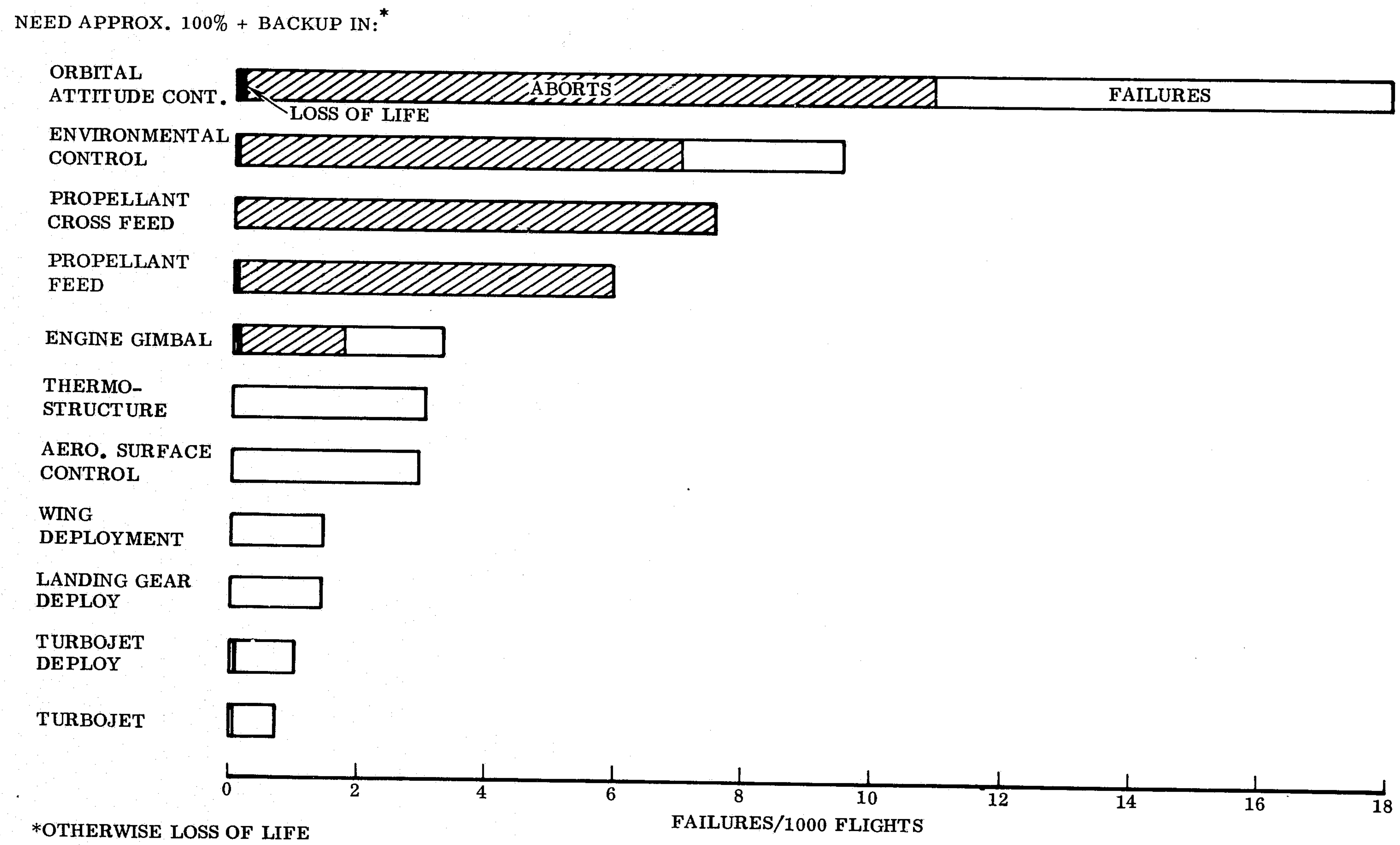


Figure 5-14. Design Improvements

5.3.7 FINAL DATA (MECHANICAL FAILURES). Figure 5-15 shows total failures occurring in each mission phase and is interpreted as follows:

a. Total failures of the baseline design include white, shaded, and black bars. The total = 84.9/1000 flights.

b. Incorporation of recommended improvements and minor failure eliminates white bars and failures become 35.6 aborts (shaded) and 0.6 catastrophies (black bars).

The effect of incorporation of the recommended improvements is:

a. Catastrophic failures are reduced to 0.6.

b. Abort failures (mission abort) become 35.6 giving a mission reliability of > 0.96.

Mission Phase	Total Failures	Aborts	Fatalities
Liftoff to Staging	23.3	10.4	0.11
Staging to Orbit	8.5	6.8	0.03
Orbit	34.5	18.4	0.35
Entry	12.7	0	0.02
Flyback	4.3	0	0.10
Approach and Landing	1.6	0	0
Total			
Failures	84.9		
Abort		35.6	
Fatalities			0.61

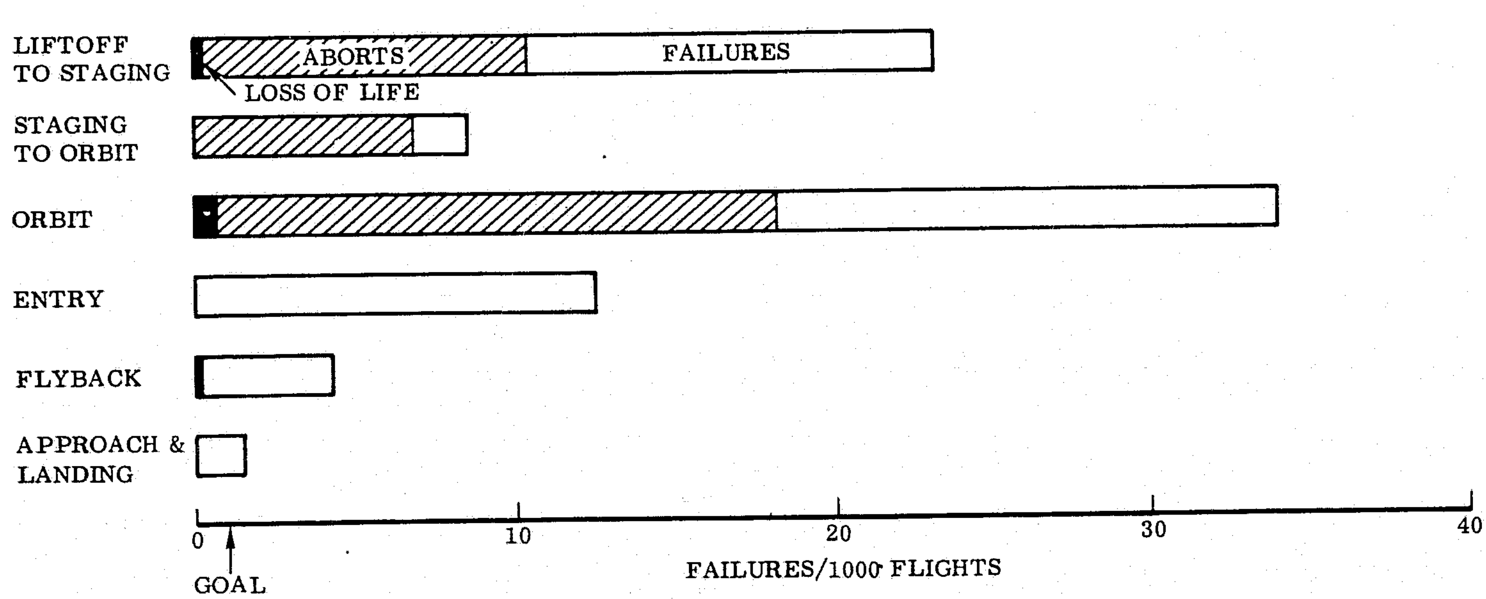


Figure 5-15. Summary of Mechanical Failures

Emphasis in design improvements should be placed on providing backup systems in several areas. Achievement of these design changes coupled with recommended abort procedures discussed earlier will result in vehicles with crew safety, intact abort, and mission success values of:

Crew safety:	0.994
Intact abort:	0.994
Mission success:	0.964

5.3.8 LOSSES DUE TO FIRE AND EXPLOSION. An estimate of losses due to fire and explosion calculated by applying a weighting factor of 30 to historical data to account for the increased hazard of a H_2/O_2 propellant compared with JP-4 fuel will yield 0.15 losses/1000 flights as follows:

Convair report, GDC-64-273, Investigation and Study of Fire and Explosion Hazards for F-105 airplanes, shows F-106 losses due to fire and explosion to be 0.005/1000 flights. This reports points out deficiencies in the design approach used in the F-105 and shows how good design can reduce losses. This approach has been demonstrated in the F-106, where losses are reduced by an order of magnitude by good design practices.

With this rational applied, losses due to fire and explosion can be held to 0.15/1000 flights. This value is excessive and points up the need to design for safety (along the lines of the F-106 design approach used in the above mentioned report) to reduce this hazard so that losses of this kind are held to essentially zero. If the problem is recognized soon enough in the design stage these losses can be eliminated. The term 0.15/1000 is used in the analysis for comparison purposes. Adding mechanical failures (0.61) gives catastrophic losses equal to 0.8/1000 flights.

5.3.9 TIME CRITICAL FAILURES. In addition to the mechanical failures discussed above, other failures of a time critical nature will occur at relatively low frequency where it will be necessary to separate prior to staging. These are cases where it is desirable to limit the kinetic energy buildup of the system because of structural or thermal protection system failure or where immediate separation and abort is required due to fire. Following separation, a throttle/burn/dump/reverse flight operation can be used. This abort procedure is described in detail in Section 5.4.

5.3.10 TIME CRITICAL ANALYSIS FOR PROPELLANT FEED. Loss of the primary propellant feed system is time critical. An analysis was made to determine the maximum time allowable for thrust interruption to prevent the trim angle from exceeding the selected limit value.

The FR-1 boost configuration was examined at max αq and near staging q to determine the effects of propellant feed failure in one booster resulting in loss of thrust to

two engines in that booster and one engine of the orbiter. High gimbal rates delay angle of attack buildup as shown in Figure 5-16. Although the vehicle is aerodynamically stable, the moment (92×10^6 ft-lb) due to thrust failure cannot be counteracted by a reasonable angle of attack.

Gimballing the three firing engines reduces the moment by 43×10^6 ft-lb. Crossfeed in the orbiter that would permit bringing the second orbiter engine back to full thrust would relieve 24×10^6 ft-lb leaving an imbalance of 25×10^6 ft-lb. However, the design objective is to provide restored thrust for all three engines so that the mission continues. Figure 5-16 shows that for max αq a reaction time of 1 sec is available for restoration of full thrust by the propellant feed system with an angle of attack of 5 deg.

5.3.11 CONCLUSIONS.

a. In event of abort, provide backup and complete the boost phase. The boosters fly back to launch site and the orbiter stage goes once around the Earth and lands at the launch site.

b. Time critical failures (structural failure or fire) require provisions for immediate separation and abort.

c. Predicted intact abort success (safety): 0.9992 (0.8 losses/1000 flights)

   Predicted mission success probability: 0.96 (35 aborts/1000 flights)

   These values compare favorably with established goals of 0.999 (1/1000) and 0.97 (30/1000), respectively.

5.4 ABORT PROCEDURES

It is necessary to develop operating procedures to employ when a failure occurs at the fail-safe level. The basic procedure for such abort situations is that the orbiter continue on to orbit and go once around the Earth and perform a glide return to the launch site. A schematic of this maneuver is shown in Figure 5-17. For the baseline 55-degree orbit the crossrange required after entry to return to ETR is about 800 n.mi.

5.4.1 ABORT DURING LIFTOFF TO STAGING. The booster and orbiter elements will complete the boost phase reaching a staging point with booster propellants depleted and orbiter propellant tanks full. The vehicles separate and the booster returns to the launch site in a normal manner while the orbiter continues once around the Earth and returns to the launch site. The once-around abort procedure reflects a high probability of successful intact abort from all failures of a mechanical nature such as engine failure or gimballing.

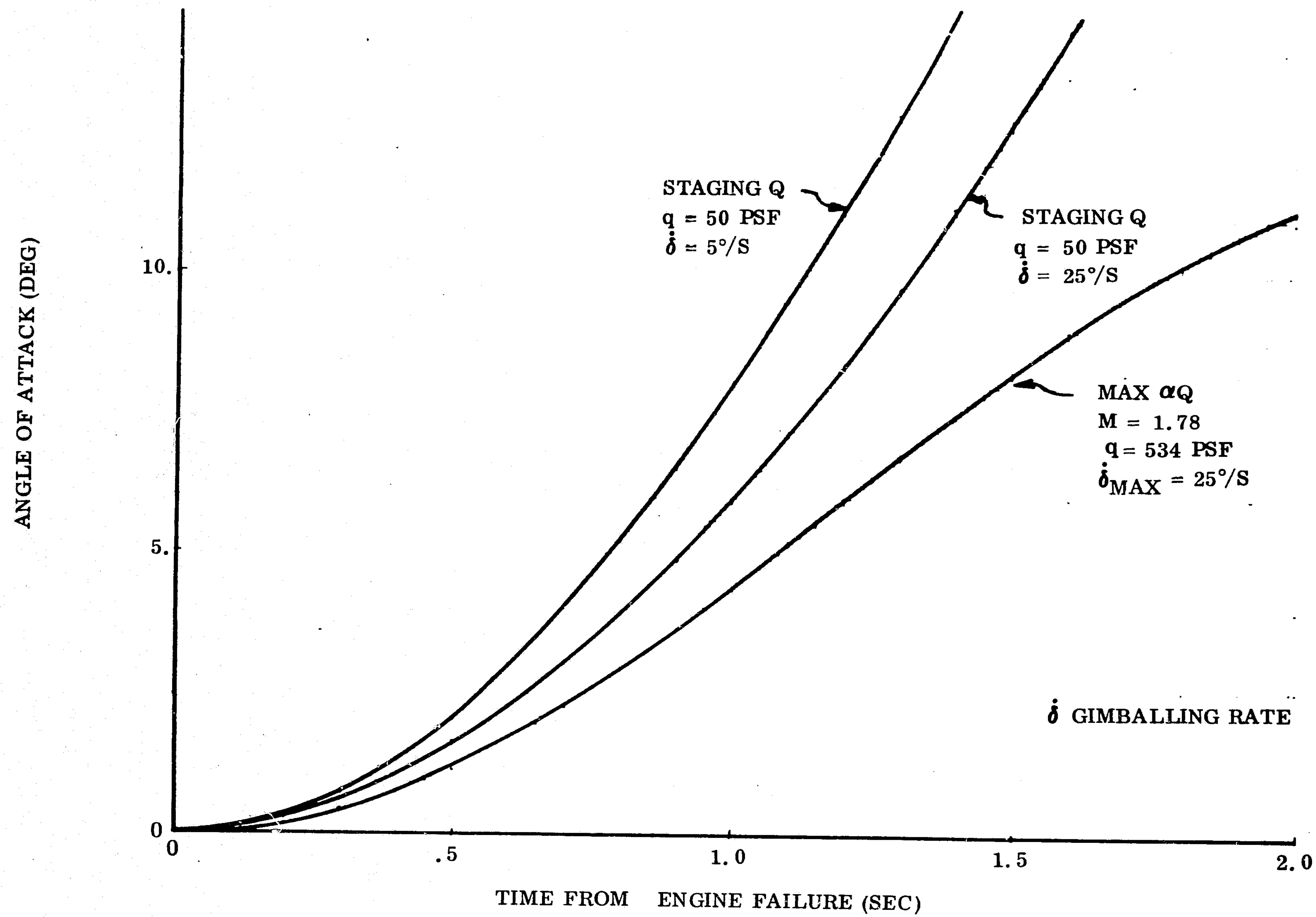


Figure 5-16. Angle of Attack Versus Time from Engine Failure

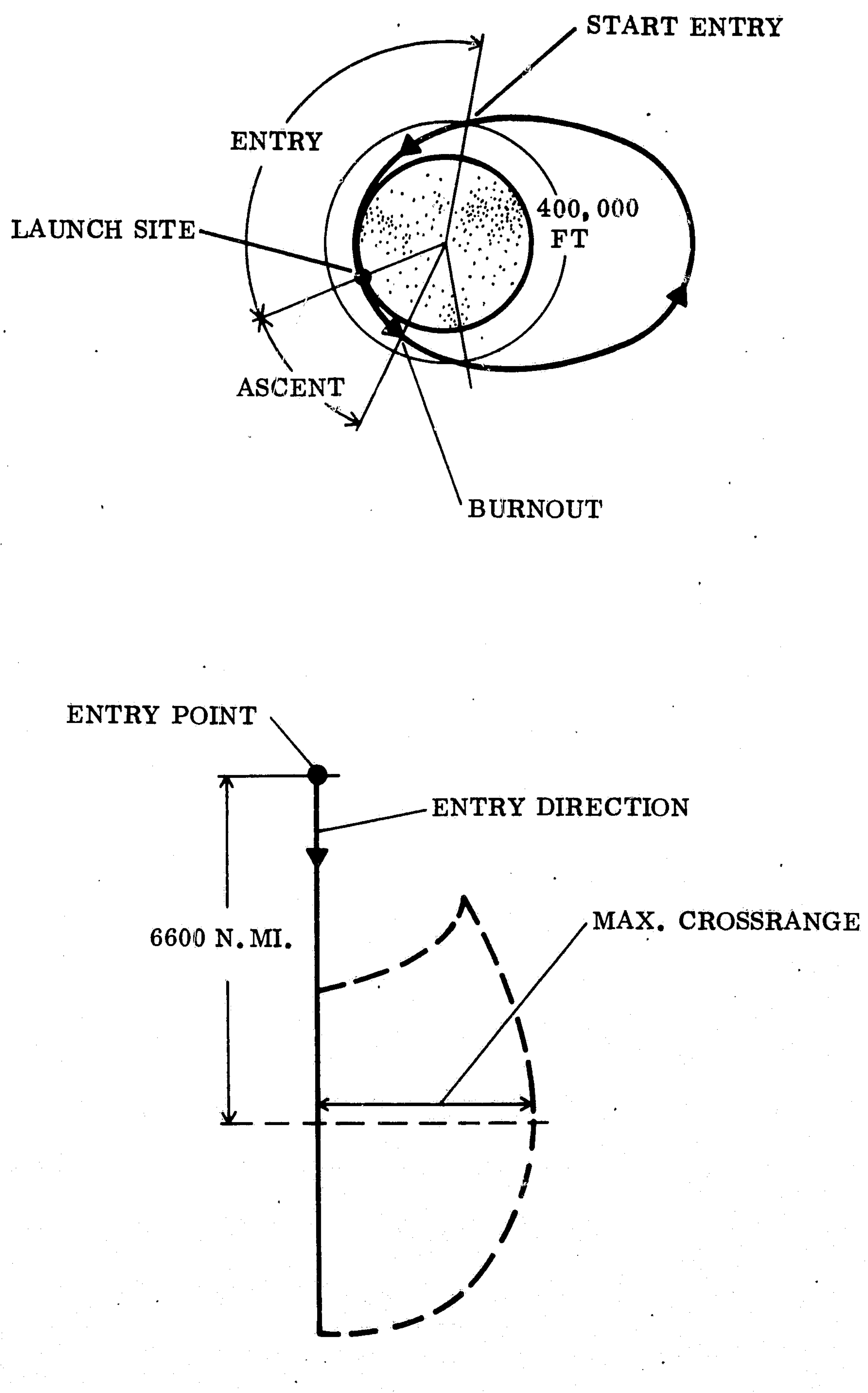


Figure 5-17. Once Around Schematic

In addition, there are rare failure situations which require immediate abort. Some typical types are structural failure, thermal protection system failure, time critical failures, hardover gimbal, and catastrophic situations such as a fire which may require early separation. On the pad, or just off the pad, the vehicles separate, climb to some suitable altitude, say 30,000 ft, and throttle back engines to burn off propellant while limiting altitude and velocity. When empty, each vehicle used an equilibrium glide path intercept to land at the launch site.

Above 30,000 ft altitude (approximately Mach 1.0) the procedure is similar in that the booster and orbiter separate and deplete their propellants through the rocket engine. However, the vehicles are downrange and have high velocity. If the vehicles continue thrusting, both booster and orbiter are liable to be too far downrange to be able to return to the launch site. This is particularly critical for the orbiter because it has very little subsonic flyback range. Rocket propellants and rocket thrust are therefore used to return the vehicles to the vicinity of the launch site.

This reverse maneuver was investigated to establish that the orbiter was in fact capable of returning directly to the launch site. Figure 5-18 presents a typical trajectory in terms of altitude and outgoing or return velocity. The critical case in terms of performance occurs at staging. From a nominal staging velocity of about 8000 feet per second at a 10.9-degree flight path angle, the orbiter is assumed to apply its main propulsion in a retro rocket fashion at some positive attitude (0 to 10 deg) to nullify forward velocity and develop sufficient velocity towards the launch site to be able to continue on an equilibrium glide trajectory back to the launch site. An upward direction of thrust is required in order to have this maneuver intercept an equilibrium glide trajectory. For a glide at maximum L/D, the point indicated in Figure 5-18 represents the case when the orbiter range out during boost and retro to zero velocity is equal to the range back during acceleration to and gliding on an equilibrium path. Total velocity increments on the order of 14,000 to 15,000 feet per second are required for this maneuver, well within the ΔV capability of the orbiter at staging.

5.4.2 ABORT AFTER STAGING. For a critical failure occurring in the orbiter, after staging and prior to injection, the abort procedure is to switch to a backup mode, continue once around the Earth, and return to the launch site. During orbit or entry the procedure is to switch to a backup mode and return to the launch site.

5.4.3 PROPELLANT DUMPING. A gross analysis of the requirement for propellant dumping has established that the need for dumping is limited to special cases of low frequency of occurrence. In event of a structural or thermal protection system failure it is desirable to limit energy buildup (velocity) and in event of a time critical failure (fire) it is desirable to terminate boost (abort).

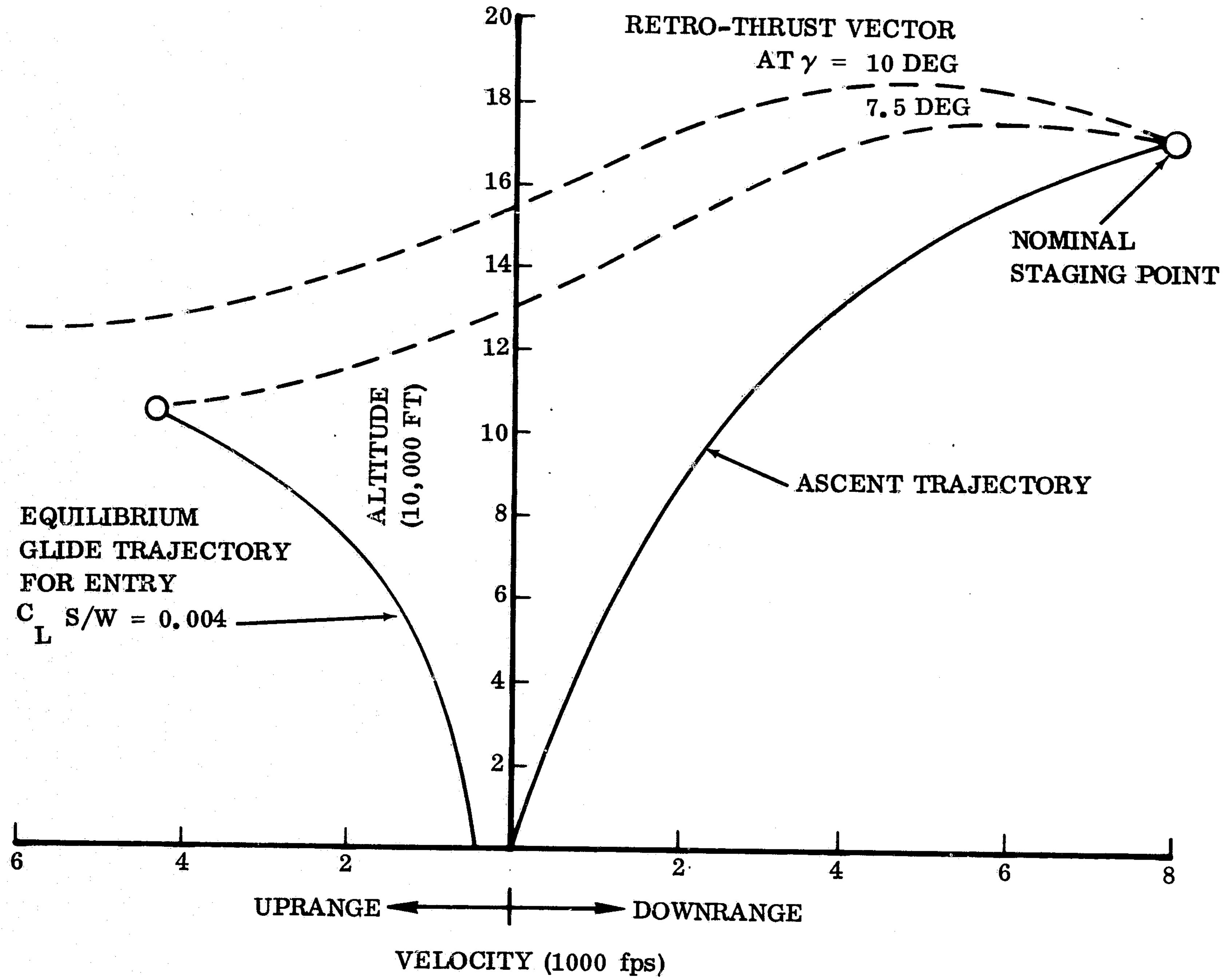


Figure 5-18. Orbiter Reverse Maneuver

Problems associated with accomplishing actual dumping involve (1) high dump rates due to short time of onset of excessive load factor and heating, (2) adverse gravity axis (zero g and aerodynamic drag); and (3) reduced reliability of propellant feed system due to complexity of providing propellant dumping.

Investigation of the abort situation following dumping reveals that for most launch azimuths landing sites are not available for the orbiter due to lack of flyback capability. Ditching is not considered a satisfactory solution because of cost, even if the orbiter vehicle incorporated the penalties required for a safe water landing. Vehicle losses must be held to $< 1/1000$ to make the reusable concept feasible.

5.5 DESIGN REQUIREMENTS

Vehicle and vehicle system design requirements which will enable accomplishment of the abort procedures specified in Section 5.4 are:

Following failure of:

a. Any rocket engine:
   1. Design to keep going.
   2. Design liftoff thrust/weight $\geq$ 1.16
   3. Design orbiter to fly once around the Earth and land at launch site.

b. Propellant feed:
   1. Provide backup in 1 second (propellant feed tank pressurization).
   2. Design propellant feed backup for full thrust capability.
   3. Design to deplete boosters equally before staging (for FR-4 by throttling opposite booster engines).
   4. Design for once-around mission:
      a) Use main propellants.
      b) Use or dump orbit maneuvering propellant.

c. Thermostructure
   1. Design for any engine out in any part of the trajectory.
   2. Design for any turbojet out during flyback.
   3. Design for failure transients.
   4. Provide redundant attachments, etc.
   5. Provide multi-load path design.

d. Subsystem. Design for fail operational/fail safe operation. Design provisions such that after failure of:

Subsystem	Action
Guidance Power Hydraulics Electrical ECS ACS Avionics	Switch to backup. Complete mission. After second failure, return to launch site.
Gimbal Aero Control Surfaces Wing Deploy Turbojet Deploy Landing Gear Deploy Inerting Systems	Perform function, may be at reduced rate. Complete mission. After second failure, return to launch site.

5.6 DESIGN FOR SAFETY (FIRE AND EXPLOSION HAZARD)

5.6.1 INTRODUCTION. The hazard of fire and explosion on the FR-1, FR-3, and FR-4 vehicles is potentially greater than on a conventional JP fueled aircraft because of the large quantity of H_2, which has a very broad flammability range and a low ignition energy. There is always the possibility that a leak somewhere in the LH_2 system will generate gaseous hydrogen. For the ascent phase of flight and during entry, ambient air containing oxygen can enter the interstitial space where GH_2 could form.

Since the concentration of GH_2 required to form a combustible mixture has a broad range (4% to 75% by volume), it must be assumed that a small cryogenic leak of LH_2 could subsequently vaporize and form a mixture whose concentration is within the flammable range. The gaseous mixture is not hypergolic. However, because of the low ignition energy required (0.019 millijoules, 1/10 that for JP fail and air) it must be assumed that an ignition source in the form of electrical spark (chaffed wires/or static discharge) is always present. Since sources of ignition in accidents reported (except Apollo) have not been identified conclusively, no potential source can be disregarded. During entry, stratified layers of hot air above the spontaneous ignition temperature could enter the interstitial space and ignite any combustible mixture that may be present.

Table 5-3 shows a comparison of the relative hazard of a reusable launch vehicle and an airplane. It shows that hazard is greater for the reusable vehicle because of the type of propellant, the amount of propellant, and the vehicle size compared with an F-106 airplane. Figure 5-19 shows schematically the hazard potential (sources of fire and explosion) in the vehicle for an early FR-1 design. The vehicle carries quantities of LO_2, JP-4, and LH_2, the major source of fire and explosion hazard.

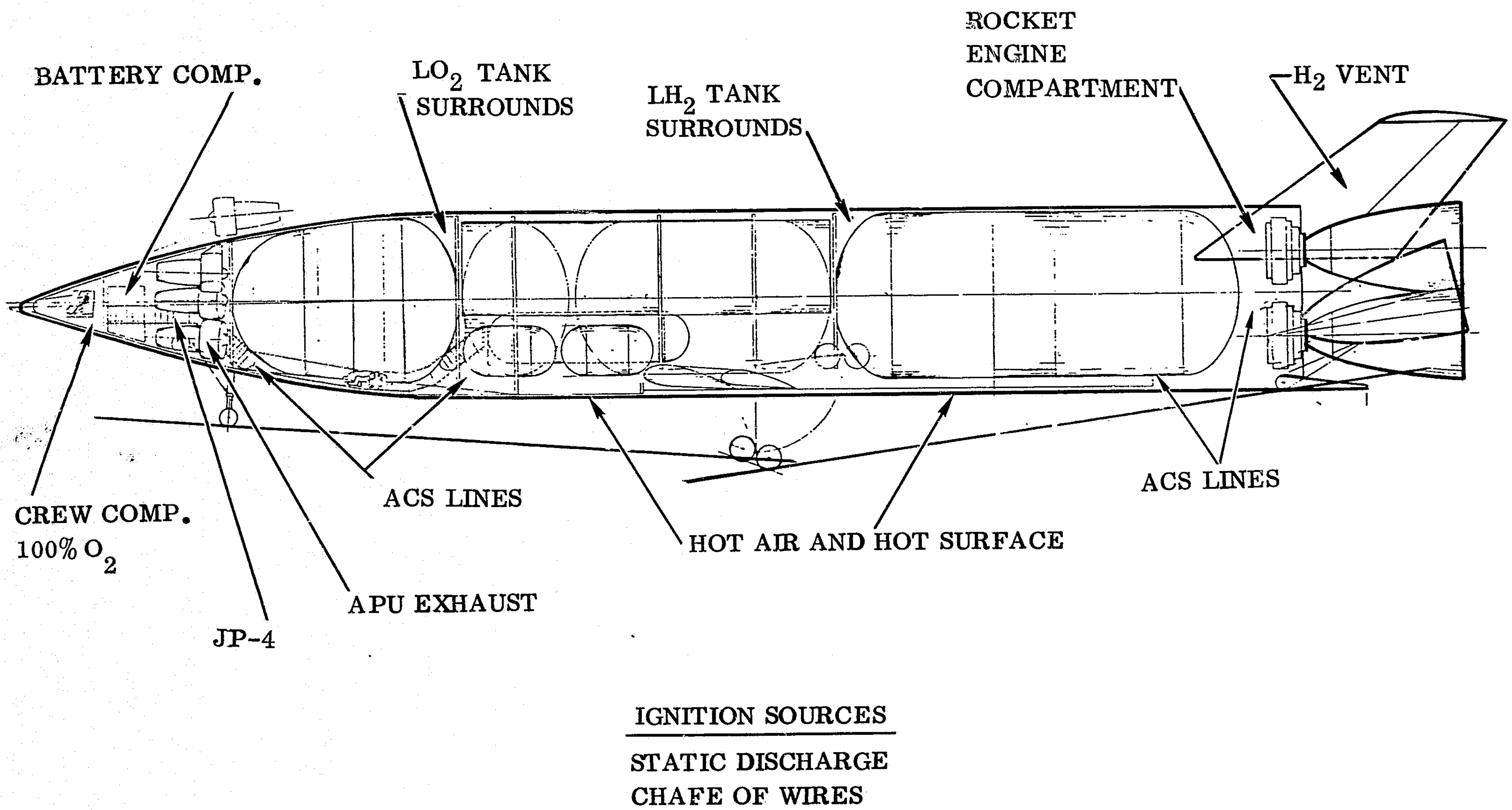


Figure 5-19. Hazard Potential

Table 5-3. Hazards Comparison — Reusable Launch Vehicles/Airplanes

•Propellant and Fuel Characteristics (1 Atmos.)	H_2/O_2	H_2/Air	JP-4/Air
Flammability Range (% by Volume)	4-94 (520°R)	4-75 (520°R)	0.7-4.8 (520°R)
Ignition Energy Required (Millijoules)	0.003	0.019	0.2

•Vehicle Characteristics		
Propellant/Fuel	Reusable Launch Vehicle* (H_2/O_2 Propellant)	Aircraft (F-106) (JP-4 Fuel)
Weight at T.O.	≈2 1/2 Million lb	9,018 lb
ρ (Density)	At mixture ratio 7:1 average density = 24 lb/ft^3	48.5 lb/ft^3
Volume	≈100,000 ft^3	186 ft^3

*Early FR-1 vehicle.

Figure 5-20 shows schematically the design safety requirements. These design provisions are necessary in order to demonstrate that the vehicle will be operationally safe, and to ensure that failures due to fire and explosion do not occur.

In summary, the vehicle design requires the following in order to eliminate occurrence of fire and explosion:

a. Provide sealed, gas tight bulkheads to separate compartments containing fuels and/or oxidizers.

b. Provide diaphragms to seal off hot air and isolate hot surface ignition sources.

c. Provide purging with an inert gas during ascent and descent to an O_2 concentration < 2% by volume for LH_2 tank surrounds, the rocket bay, and the payload bay.

d. Provide O_2/N_2 crew and passenger compartment atmosphere.

e. Apply design practices to eliminate leakage potential and isolate electrical ignition sources.

5.6.2 PURGE AND POSITIVE PRESSURE CONTROL SYSTEM

5.6.2.1 Purge of Interstitial Space. This initial assessment indicates that a solution to the fire and explosire hazard is to provide an inert gas to suppress any potential fire or explosion with a purge and positive pressure control system.

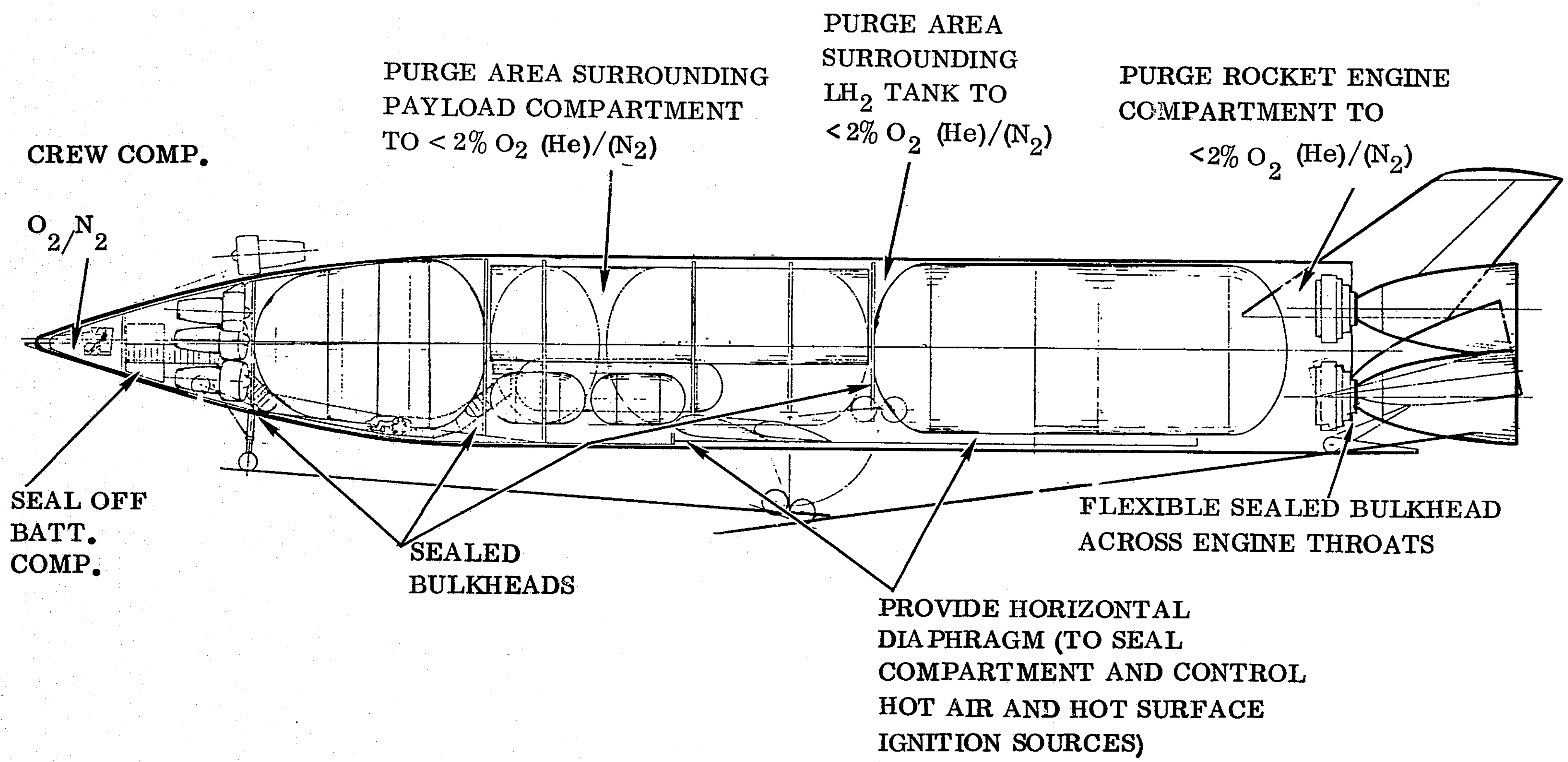


Figure 5-20. Design for Safety

The basic premise for the design evolves from consideration of several probabilities. The probability of occurrence of a small cryogenic leak of LH_2 or LO_2 is equal. The probability that both of these will occur at the same time is remote. The probability of a line breakage resulting in a large flow of either LH_2 or LO_2 is remote. The probability of ambient air being present in the interstitial space is a certainty unless a positive pressure is maintained in the space so that ambient air containing O_2 cannot enter. If the pressurizing gas in inert and if the O_2 concentration < 2% by volume, combustion is not possible even with a large concentration of H_2. The potential of fire or explosion from this cause is thus removed. The system consists of:

a. Provisions for introducing an inert gas (N_2) into the rocket engine bay, the area surrounding the LH_2 tank, and the area surrounding the payload bay on the ground.

b. Pressure regulator and vent valves, one of which is shown schematically in Figure 5-21, located in these same spaces to control interstitial space pressure to a positive value above ambient and within structural limits during ascent. The valves are open in orbit and close during entry and descent.

c. Stored inert gas which is used to repressurize the interstitial space during entry and descent.

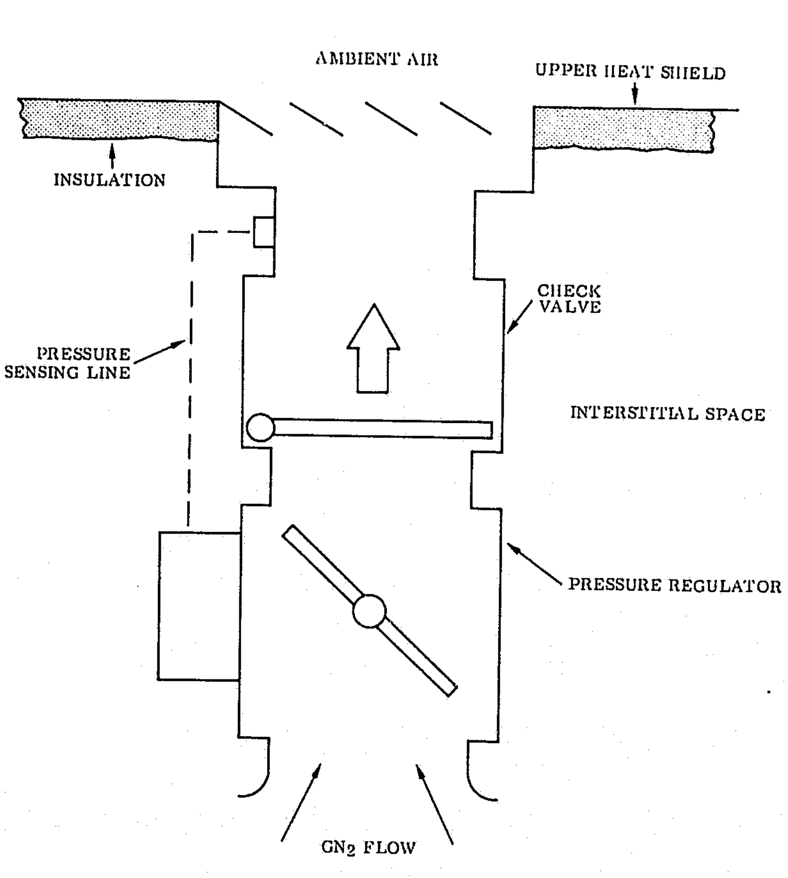


Figure 5-21. Pressure Regulator Vent and Check Valve Schematic (PRV)

The LH_2 tank surrounds, the rocket engine, and the payload bay are purged and pressure-controlled. The LO_2 tank surrounds and turbojet bay are not because these contain no H_2. These bays are pressurized with ambient air and are provided with a lower pressure barrier on the cold side of the lower heat shield which prevents hot air from entering the compartments. The small amount of heat transfer from air from the upper heat shield area used to repressurize these compartments on entry is negligible.

The LH_2 tanks themselves cannot be purged during ascent, when the probability of failure which might lead to fire or explosion is greatest. For the orbiter, residual gaseous hydrogen and gaseous oxygen is used for the ACS and APU operations. Time does not permit purging of LH_2 and GH_2 from the main tanks for booster or for the orbiter under an abort condition.

5.6.2.2 Purge and Pressure Control. The basic pressure control schedule is shown in Table 5-4.

5.6.2.3 Source of Repressurizing Inert Gas (He). Gaseous He is stored in a container inside the long term LH_2 tanks at 30°R. A system consisting of a gas generator and a heat exchanger is used to expand the gas for efficient volume utilization.

Table 5-4. Purge and Pressure Control Schedule

Trajectory Phase	Interstitial Space (I/S) Pressure	Function Performed	System Operation
Ground	$\Delta P = +1$ psig with GN_2	Compartment is purged to $[O_2] < 2\%$ by volume. Positive pressure keeps ambient air out of I/S.	Pressure regulator/vent valve (PRV) holds pressure to 1 psig nominal.
Ascent	$\Delta P = +1$ psig with GN_2. (GN_2 is venting overboard)	$[O_2] < 2\%$. Positive pressure keeps ambient air out.	PRV controls pressure to structural limit.
Orbit	I/S vented to space — hard vacuum	This gets rid of fuel and oxidizer gases, which if retained would represent a hazard. Provides safe condition for orbital stay.	PRV open to vacuum.
Pre-entry and Entry	I/S repressurized with inert gas (He) to $\Delta P = 1$ psig	Positive pressure keeps ambient air out	PRV closed (electrically) pressure is controlled by inflow regulator in He supply system.
Descent,	I/S repressurized with inert gas (He) to $\Delta P = 1$ psig	Positive pressure keeps ambient air out	PRV closed (electrically) A 10 min. post landing provision provides for time to connect ground cart.

I/S = Interstitial Space
[ ] = Concentration

5.6.2.4 Pressure Regulator Valve. During ground purge, the valve holds pressure to $\Delta P = 1$ psi. During ascent, purge gas is vented at a flow rate sufficient to hold pressure < 1 psi above ambient. In orbit, the valve is opened and all residual gases vent to vacuum. During entry the valve is closed, and compartment pressure is controlled by an inflow regulator which is part of the GHe supply system. Four 18-in. diameter (flow area) pressure regulator valves accommodate the LH_2 bay tank surrounds, providing redundancy. Any two valves can handle the vent flow required during ascent. Four are used for the payload bay, making a total of eight valves.

5.6.3 ESTIMATED WEIGHT AND VOLUME. First order estimates of the weights of the elements that provide protection against fire and explosion are:

		Orbiter	Booster
a.	GHe		
	For entry (leakage + repressurization)	220	-
	For descent (leakage + repressurization)	700	700
	For 10 min. post landing (leakage)	225	-
	For flyback boosters (leakage + repressurization)	-	1300
	For 10 min. post landing boosters (leakage)	-	225
	Weight GHe Total	1145	2225
	GHe Storage Container Weight = 1.7 × Wt GHe (usable gas + container)	1946	3782
b.	Gas Generator and Heat Exchanger	20	20
c.	8 pressure regulator valves at 20 pounds each	160	160
	Total, lb	2126	3962

The sensitivities of these weights are shown in Vol II Section 10.0.

The volume of gaseous He required is 180 ft^3 350 ft^3

5.6.4 LEAKAGE AREA. Leakage of purge gas interfaces with the pressure barriers and the heat shields. Vol. 5 Section 8 specifies leakage allowable limits for these areas. In calculating the total leakage areas for the LH_2 tank bay, rocket engine bay, and payload bay, the following value was used:

$$CA = 0.04 \text{ ft}^2$$

where

C = discharge coefficient = 0.6

A = geometric area

5.7 ENGINE EFFECTS ON SAFETY AND MISSION SUCCESS

5.7.1 INTRODUCTION. A study was conducted to determine the effect of rocket engines on safety and mission success for the FR-3 and FR-4 vehicles. The objectives were:

a. Determine safety, reliability, and mission success of the vehicles for given engine arrangements.

b. Determine the advantages and penalties associated with the application of fail-operational/fail-safe to boosters and orbiters.

c. Make a recommendation for final vehicle configurations.

Mission success and safety are a function of engine reliability, number of engines, catastrophic failure rate of engines, vehicle design in terms of thrust to weight at liftoff and thrust to weight at staging to enable engine out operation, and reliability of other subsystems not associated with engines.

In this study, subsystem failures not associated with engines are ignored and consideration is given to the engine portion of the propulsion system during the ascent phase. Failure modes associated with engines are identified as follows:

a. Catastrophic failures (engine catastrophic failures resulting in loss of adjacent engine and loss of vehicles).

b. Loss of performance (thrust loss is non-catastrophic).

A mission success goal of 0.97 (30 aborts/1000 flights) and a crew and passenger safety goal (intact abort) of 0.999 (1 loss/1000 flights) were established for the study. These goals required that an assessment of the potential catastrophic engine failure be made particularly since use of the 400,000-lb engine requires more total engines than use of larger engines. The probability of catastrophic failure is directly proportional to the number of engines, while performance failure (loss of thrust which may negate abort capability) diminishes with increasing numbers of engines.

Further, as the vehicle advances along the boost trajectory, fewer and fewer numbers of engines are required to accomplish a safe return. As less and less engines are required, the probability of vehicle losses due to engine performance failure approaches zero.

Three engine reliabilities — 0.99, 0.997, and 0.999 — were studied. These reliabilities apply to the total burn time of 0.155 hr. The liftoff to staging (boost 1) and staging to orbit (boost 2) times were taken as equal at 0.06 hr. Orbital operations were not included in the trade study since the minimum number of orbiter engines studied was three which provide fail operational/fail-safe operation for on-orbit maneuvers.

The fail-safe and fail-operational/fail-safe concepts were applied to boosters and orbiters to different degrees, and the penalties for achieving different reduced loss rates (missions and vehicles) were determined.

Reliability considerations following one engine out, two out, and three out are:

a. Can the mission be completed?

b. Can a successful abort be accomplished?

In a given engine vehicle configuration, if the mission can be completed with an engine out and a successful abort made with two engines out, this is equivalent to saying the vehicle system is capable of a fail-operational/fail-safe operation. If the mission cannot be completed with one engine out, but safe return is possible, the vehicle is capable of fail-safe operation. The minimum mission is safe return of the vehicle with at least one engine out. Specific values of liftoff and staging thrust to weight ratio must be maintained to achieve successful fail-safe and fail-operational/fail-safe. In addition, overthrust capability must be provided to allow fail-operational/fail-safe performance.

5.7.2 PROBABILITY OF ENGINE CATASTROPHIC FAILURE. A quantitative assessment of the probability of engine catastrophic failure is not available, though Pratt and Whitney has estimated that 1% of all engine failures will be catastrophic resulting in loss of the vehicle. Vehicle losses must then include consideration for these catastrophic engine failures. The more engines there are, the higher the probability of vehicle loss due to catastrophic failure. The bell and aerospike engines were investigated to determine the characteristics of catastrophic failure modes.

Mission reliability is greatly improved when the fail-operational/fail-safe concept is used instead of fail-safe, but vehicle losses remain essentially unchanged because of the catastrophic ratio. The major factor in engine design is the reduction of the catastrophic failure ratio.

The bell engine has a set of turbine-driven propellant pumps, a pre-burner chamber and a thrust chamber and nozzle for each engine element.

5.7.2.1 Design Features Involving Catastrophic Failure Considerations. The basic cycle of the bell high pressure rocket engine has the following design features:

a. Mixture ratio control (first level sensor).

b. Closed loop turbine cycle.

c. Preburner temperature sensor (second level sensor).

d. Transpiration wafer pressure sensor (second level sensor).

e. Ten hour life and reuse requirement.

Features a, b, c, and d tend to turn down thrust if variation of propellant flow occurs for any reason. The probability of cracks occurring in heat exchangers and transpirtation wafers is reduced because of the requirement for reuse which calls for greater design margins (reduced stresses and over cooling).

Temperature and pressure sensors will act as second level monitoring sources to shut down thrust as required to prevent catastrophic failure which may damage adjacent engines or the spacecraft.

5.7.2.2 Failure Modes

a. Turbine Bearing Freeze. The engine controls mixture ratio so that reduced fuel or oxidizer flow will turn down thrust. Reduced fuel flow will increase the mixture ratio but before the preburner temperature reaches stoichiometric values, the preburn temperature sensor will shut down the engine.

b. Turbine Overspeed. The closed loop turbine cycle tends to prevent turbines from overspeeding because of choked flow at the nozzle throat. As propellant flow (gas) is increased, the turbines are back pressured and will slow down or reach a limit speed because of reduced pressure ratio. This would not be the case if a separate gas generator is used such as in the Atlas engine. The turbines handle large mass flows (large turbines) and operate at low pressure ratio. Overspeed of turbine can occur with a sudden reduction in source flow of fuel or oxidizer because energy is momentarily stored in the preburner case, which has a relatively large volume.

c. Propellant Flow Interruption (Gradual)

   For oxidizer: assuming reduced flow upstream of inducers, if NPSH is lost or the closed prevalve, if used, fails the mixture ratio control will turn down thrust.

   For fuel: with the same assumptions as above, the mixture ratio control (first level sensor) will turn down thrust, and the preburner temperature sensor (second level sensor) will shut down the engine.

d. A crack in the heat exchanger (bell nozzle) will dump fuel tending to increase the preburner mixture ratio. As the stoichiometric temperature is approached the preburner temperature sensor will shut down the engine before the preburner case fails.

e. A crack in the transpiration wafer will tend to increase flow in the area of the crack which will reduce H_2 flow to the preburner. The preburner temperature sensor will turn down the thrust as the temperature reaches a set limit. Wafer pressure will also act to shut down the engine.

It is recommended that further work be done by this engine manufacturer on a failure analysis. The basic objective in the failure analysis should be to make compromises in favor of a fail-safe design (i.e., an engine which does not fail catastrophically in a way which may damage adjacent engine or the spacecraft). During the test program, temperature and pressure spikes which could cause failure should be carefully monitored.

5.7.2.3 Aerospike Engine. The basic engine consists of a set of turbine-driven propellant pumps, a preburner chamber and a nozzle for each set of 400,000 lb elements.

<u>Design Features Involving Catastrophic Failure Considerations</u>. The basic cycle of the aerospike rocket engine has the following features:

a. Mixture ratio control (first level sensor).

b. Open loop turbine cycle.

c. Preburner temperature sensor (second level sensor).

d. Ten-hr life and reuse requirement.

Features a and c tend to turn down thrust if variation of propellant flow occurs for any reason. As in the bell engine, the probability of cracks occurring in the heat exchangers is reduced because of the requirement for reuse, which calls for greater design margins (reduced stresses and over cooling).

Temperature and pressure sensors will act as second level monitoring sources to shut down thrust as required to prevent catastrophic failure which may damage adjacent engines or the spacecraft.

<u>Failure Modes</u>

a. Turbine Bearing Freeze. The engine controls the mixture ratio so that reduced fuel or oxidizer flow will turn down thrust. Reduced fuel flow will increase the mixture ratio but before the preburner temperature reaches stoichiometric values, the preburn temperature sensor will shut down the engine.

b. Turbine Overspeed. The open loop turbine cycle is in effect a gas generator with turbine discharge pressure essentially ambient. The turbines handle a relatively small mass flow (smaller turbines) and operate at high pressure ratio. Although overspeed of turbines can occur with a sudded reduction in source flow of fuel or oxidizer, it is not likely to because of the reduced turbine size and less stored energy in the smaller preburner.

c. Propellant Flow Interruption (Gradual)

   For oxidizer: assuming reduced flow upstream of inducers, if NPSH is lost or the failed closed prevalve, if used, fails, the mixture ratio control will turn down thrust.

   For fuel: with the same assumptions as above the mixture ratio control (first level sensor) will turn down thrust, and the preburner temperature sensor (second level sensor) will shut down the engine.

d. A crack in the heat exchanger will dump fuel tending to increase the preburner mixture ratio. As the stoichiometric temperature is approached the preburner temperature sensor will shut down the engine before the preburner case fails.

It is recommended that further work be done by this manufacturer on a failure analysis. The basic objective in the failure analysis is the same as for the bell engine: to make compromises in favor of a fail safe design (i.e., an engine which does not fail catastrophically in a way which may damage adjacent engine or the spacecraft). During the test program, temperature and pressure spikes which could cause failure should be carefully monitored.

5.7.3 MINIMUM NUMBER OF ENGINES REQUIRED FOR SAFETY

5.7.3.1 Fail Safe Approach. For some portion of the boost 1 phase, only one engine failure can be tolerated for the "once around" abort to be still accomplished. But for the remainder of the boost 1 phase, two, three, and even more engine failures can be tolerated and the abort can be accomplished. In other words engines may fail as the vehicle advances along the trajectory and a safe return still can be made.

If this fact is taken into account, vehicle loss calculations will show significantly reduced values because the time required for n-1, n-2, and n-3, etc., engines operating is drastically reduced. Vehicle losses beyond n-3 engines are negligible.

One approach to assessment of the quantitative effect of this is to assume that the time required for n-1, n-2, n-3, etc., engines operating is a function of n (number of engines).

If time required for n-1, n-2, etc., engines operating is some function of total trajectory (boost 1 phase) time divided by number of engines, then:

$$t_{\text{req'd for n-1, n-2, etc., engines operating}} = \frac{t_{\text{traj phase total time}}}{n} \qquad (1)$$

Equation 1 was modified in the interest of conservatism by the application of a factor of 2, giving the expression used to calculate losses:

$$t_{\text{req'd for n-1, n-2, etc., engines operating}} = \frac{(2)\ t_{\text{traj phase}}}{n} \qquad (2)$$

5.7.3.2 Fail-Operational/Fail-Safe Approach. A similar situation is applicable to the fail-operational/fail-safe approach, but here vehicle safety is further enhanced because the added engines reduce the probability of having less than the number of engines required for a "once around" abort.

The effect of providing fail-operational/fail-safe to improve mission reliability then also improves vehicle safety by the application of Equation 1 if the engine catastrophic failure ratio is $\ll 1\%$.

5.7.4 IMPLICATIONS OF UPRATED THRUST RATING

5.7.4.1 Overthrust (Overspeed) of Engines. The propellant utilization (PU) system significantly reduces residual propellants which improve vehicle performance. This system monitors tanked mixture ratio and regulates engine mixture ratio to give full utilization of available propellant (simultaneous depletion of fuel and oxidizer). The rocket engine is designed to provide a range of mixture ratios for this purpose. The engine so designed will actually have an increased thrust capability at some fixed mixture ratio. This is because of the pump and thermal limits required in the engine design. This situation is shown in Figure 5-22. For the 400,000-lb engine currently defined by Pratt and Whitney, this capability is 108 percent of nominal.

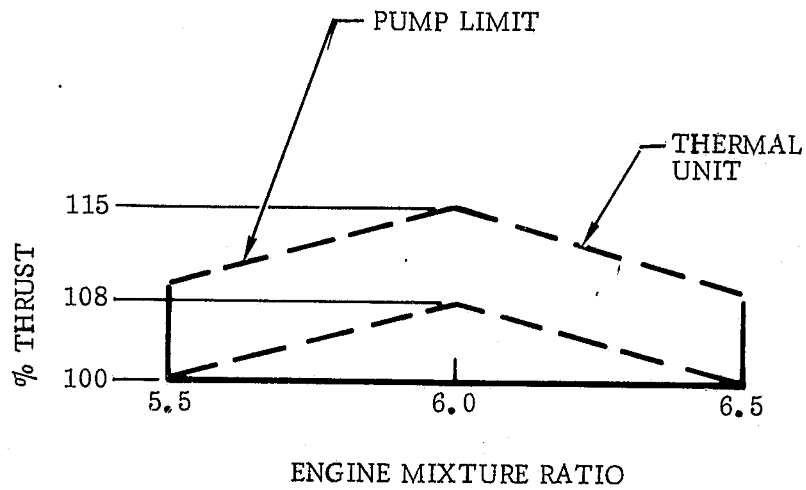


Figure 5-22. Engine Thrust vs. Mixture Ratio

Some overthrust capability is then inherent in the engine design. However, this inherent overthrust capability cannot be used for fail-operational purposes without significant performance penalty since no PU control is possible. To provide PU control capability at thrusts above nominal, the engine must be designed to operate beyond the nominal speed and temperature limits. Overthrust to 115 percent of nominal as defined in Figure 5-22 is thought to be feasible for a single flight by Pratt and Whitney. This capability would allow control of engine mixture ratio over the full design range at thrust up to approximately 7 percent. If engines are operated in this mode, some refurbishment is required after use. The cost of this refurbishment requires further investigation to determine if the use of overthrust is cost effective in terms of reduced mission aborts.

The application of this overthrust to a 15 engined booster (FR-3) can be advantageous from a mission abort standpoint. For an engine MTBF (mean time between failures) = 15 ($R_{eng} = 0.99$) mission aborts during liftoff to staging caused by engines alone can be reduced from 60/1000 flights to approximately 2, if provision is made to reach staging with one engine out (fail-operational for booster engines). This is a significant gain since it can be accomplished with presently conceived engines. The amount of overthrust required is given by $1/n-1 = 1/15-1 = 7.2\%$ which is approximately within the range of mixture ratios required for operation of the PU system. The FR-3 incorporates this fail-operational provision for booster engines. The application of fail-operational/fail safe to engines in the 9-3-9 engine configuration FR-4 vehicle is advantageous from a mission loss standpoint. Losses can be reduced from 78 to 2, but an overthrust $= 1/9-1 = 1/8 = 13\%$ is required because without propellant crossfeed, the booster element with the failed engine must produce the required performance

thrust-to-weight ratio so that it uses up all of its propellant. Under the constraint of a fixed engine module thrust engine (400, 000 lb) it would be necessary to add two engines and create a 10-3-10 engine configuration with a gross liftoff weight increase of approximately 125, 000 lb.

With the application of propellant crossfeed to the FR-4 9-3-9 configuration, the amount of overthrust for booster elements would be = 1/n-1 = 1/18-1 ≈ 6%. This would allow PU control in the present engines and a saving of the ≈125, 000-pound gross liftoff weight.

5.7.4.2 Alternate Approach (Throttling). An alternate approach to operation of engines in an overthrust mode is to provide an extra engine and operate booster engines throttled. The fail-operational/fail-safe concept applied to the FR-3 15-3 engine arrangement can be accomplished by providing 16 rather than 15 booster engines and operating the 16 normally throttled to 93%. If one engine fails, the remaining 15 good engines are operated at 100% to maintain performance T/W. The throttling approach (added engine in booster) will increase gross liftoff weight by ≈ 54, 000 lb.

5.7.5 FR-3 CONFIGURATION

5.7.5.1 Analysis. Table 5-5 presents information obtained from the study. Data were generated from Figures 5-23 and 5-24 for a 15-3 engine configuration. The 15-3 configuration was selected for the case where fail-safe is applied to both booster and orbiter. The same configuration was considered where fail-operational/fail-safe was applied to the booster and fail-safe to the orbiter. Two approaches to providing fail operational/ fail safe in the booster are shown (with overthrust and with throttling). The throttling approach requires an added booster engine.

Figure 5-23 is a plot of engine reliability versus losses for the 15-3 FR-3 installation for the liftoff to staging (boost 1) phase. Five curves are shown. The probability of one and two engines being out are shown as solid lines. Catastrophic failure probability for assumed catastrophic failures of 1% and 0.1% of the basic non-catastrophic engine failures, and probability of not having the minimum number of engines required for safe return (at left) are presented. Figure 5-24 is a similar plot for the staging to orbit phase (boost 2).

Table 5-5 presents data in terms of the degree of application of fail-safe and fail-operational/fail-safe and in terms of engine reliability. Mission losses are reduced from 72 to 13.5, as fail-operational/fail-safe is applied, for R_{eng} = 0.99 and from 21.5 to 3.7 for R_{eng} = 0.997. Vehicle losses are not significantly changed. Vehicle losses are reduced by an order of magnitude for all cases of fail-safe and fail-operational/fail-safe as the catastrophic ratio decreases from 1% to 0.1%.

A discussion of Table 5-5 in terms of engine reliability (read across) follows. Looking at the F/S case, as engine reliability is improved from 0.99 to 0.997, mission losses reduce from 72 to 21.5 and vehicle losses reduce from 0.70 to 0.23 for a catastrophic

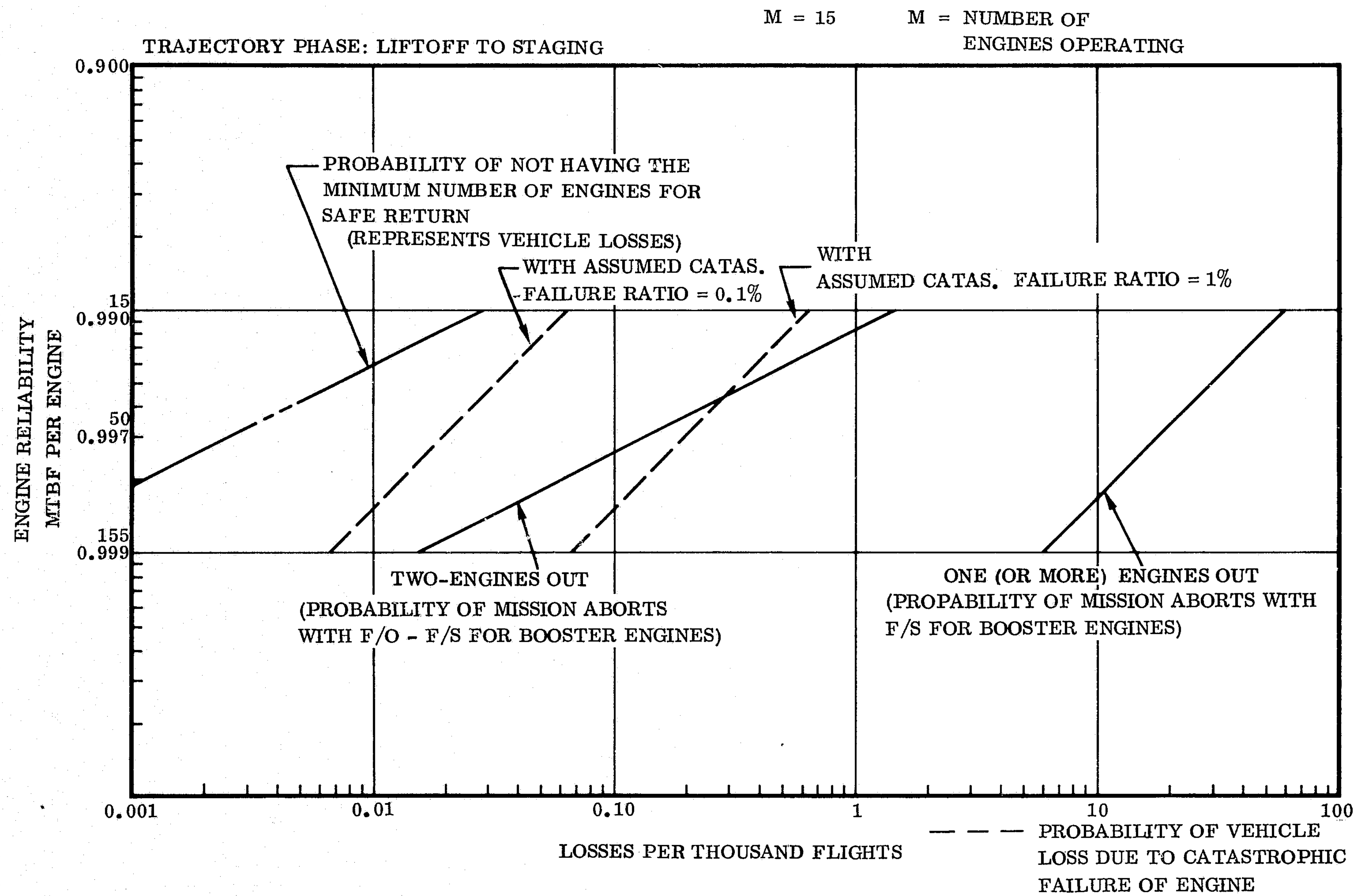


Figure 5-23. Engine Reliability Versus Aborts/Losses for 15-3 Engine Configuration

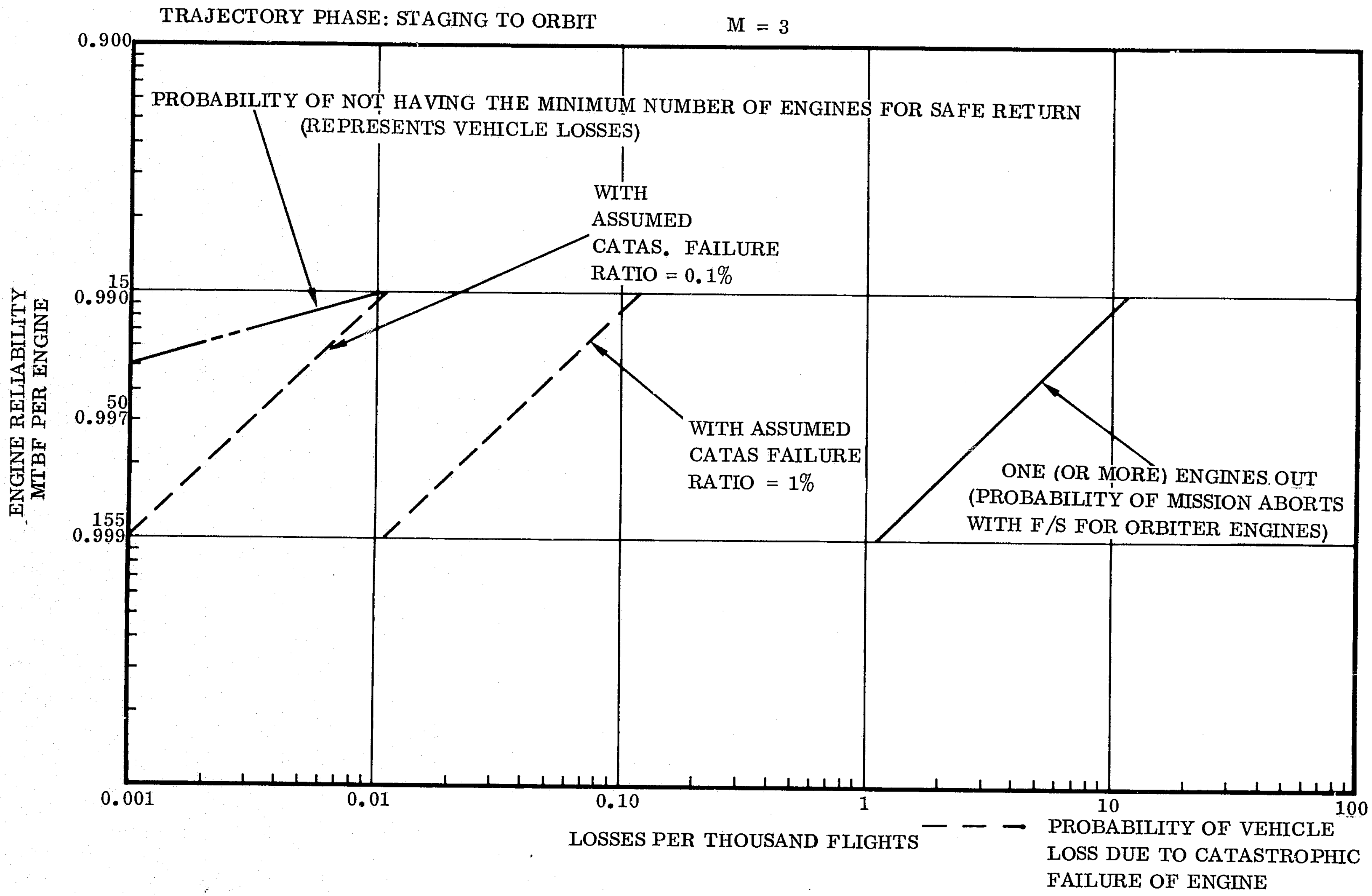


Figure 5-24. Engine Reliability Versus Aborts/Losses for the 15-3 Engine Configuration

Table 5-5. Losses/1000 Flights for 15-3 Engine Configuration

	Booster Engines		GLOW	Assumed Catas.	Missions (Abort)						Vehicles					
					$R_{eng} = 0.99$			0.997			0.99			0.997		
	Overthrust	Throttle	(M lb)	Ratio	B	O	T	B	O	T	B	O	T	B	O	T
F/S(B&O)	0%	0%	4.32	0.01	60	12	72	18	3.5	21.5	$\frac{0.6}{C}$	$\frac{0.1}{C}$	$\frac{0.7}{C}$	$\frac{0.2}{C}$	$\frac{0.08}{C}$	$\frac{0.23}{C}$
				0.001	60	12	72	18	3.5	21.5	$\frac{0.06}{C}$	$\frac{0.01}{C}$	$\frac{0.07}{C}$	$\frac{0.02}{C}$	$\frac{0.003}{C}$	$\frac{0.023}{C}$
F/O-F/S(B) F/S(O)	7%*	93%**	4.32*/ ≈4.38**	0.01	1.5	12	13.5	$\frac{0.20}{C}$	3.5	3.70	$\frac{0.6}{C}$	$\frac{0.1}{C}$	$\frac{0.7}{C}$	$\frac{0.2}{C}$	$\frac{0.03}{C}$	$\frac{0.23}{C}$
				0.001	1.5	12	13.5	$\frac{0.20}{C}$	3.5	3.50	$\frac{0.06}{C}$	$\frac{0.01}{C}$	$\frac{0.07}{C}$	$\frac{0.02}{C}$	$\frac{0.003}{C}$	$\frac{0.023}{C}$

* 15-3 Engine Configuration
** 16-3 Engine Configuration

F/O-F/S = Fail-Operational/Fail-Safe
GLOW = Gross Liftoff Weight
F/S = Fail Safe
C = Loss due to Engine Catastrophic Failure
B = Booster
O = Orbiter
T = Total

ratio of 1%, and from 0.070 to 0.023 for a catastrophic ratio of 0.1%. Looking at the F/O-F/S for booster, F/S for orbiter case, the mission improve from 13.5 to 3.7 and vehicle losses reduce from 0.70 to 0.23. This occurs because of constraints imposed by the catastrophic ratios.

5.7.5.2 Evaluation of Results. Figure 5-25 is a graphic representation of Table 5-5. This figure shows a significant improvement in mission reliability with increasing application of fail-operational/fail-safe and improved engine reliability. Vehicle safety is improved with improved engine reliability but not with application of the fail-operational/fail-safe because of the constraints of the catastrophic ratio.

5.7.6 FR-4 CONFIGURATION

5.7.6.1 Analysis. Data shown in Table 5-6 were generated from Figures 5-26 and 5-27 for a 9-3-9 engine configuration.

The 9-3-9 configuration was selected for the case where fail-safe is applied to both booster and orbiter with zero over thrust. The same configuration was considered where fail-operational/fail-safe was applied to boosters with 13% overthrust and fail safe for the orbiter.

Figure 5-26 is a graph depicting engine reliability versus losses for the 9-3-9 FR-4 installation for the liftoff to staging (boost 1) phase. Five curves are shown: the probability of one and two engines out are shown as solid lines. Catastrophic probability for assumed catastrophic failures of 1% and 0.1% of the basic non-catastrophic engine failures, and probability of not having the minimum number of engines required for safe return are presented as dotted lines. Figure 5-27 is a similar graph for the staging to orbit phase (boost 2). Table 5-6 presents data in terms of the degree of application of fail-safe and fail-operational/fail-safe and in terms of engine reliability.

In Table 5-6 mission losses are reduced from 90 to 14.2 as F/O-F/S is applied. For $R_{eng} = 0.99$ and from 24.5 to 3.7 for $R_{eng} = 0.997$ GLOW increases from 4.92 to 5.04 million pounds. Vehicle losses are not significantly changed. Vehicle losses are reduced by an order of magnitude for all cases of F/S and F/O-F/S as the catastrophic ratio decreases from 1% to 0.1%.

A discussion of Table 5-6 in terms of engine reliability (read across) follows. For the F/S case, as engine reliability is improved from 0.99 to 0.997, mission losses reduce from 90 to 24.5 and vehicle losses reduce from 0.9 to 0.23 for a catastrophic ratio of 1% and from 0.09 to 0.023 for a catastrophic ratio of 0.1%. For the F/O-F/S for booster F/S for orbiter case, mission losses decrease from 14.2 to 3.72 and vehicle losses reduce from 0.9 to 0.23. Again this is because of the constraints imposed by the catastrophic ratios.

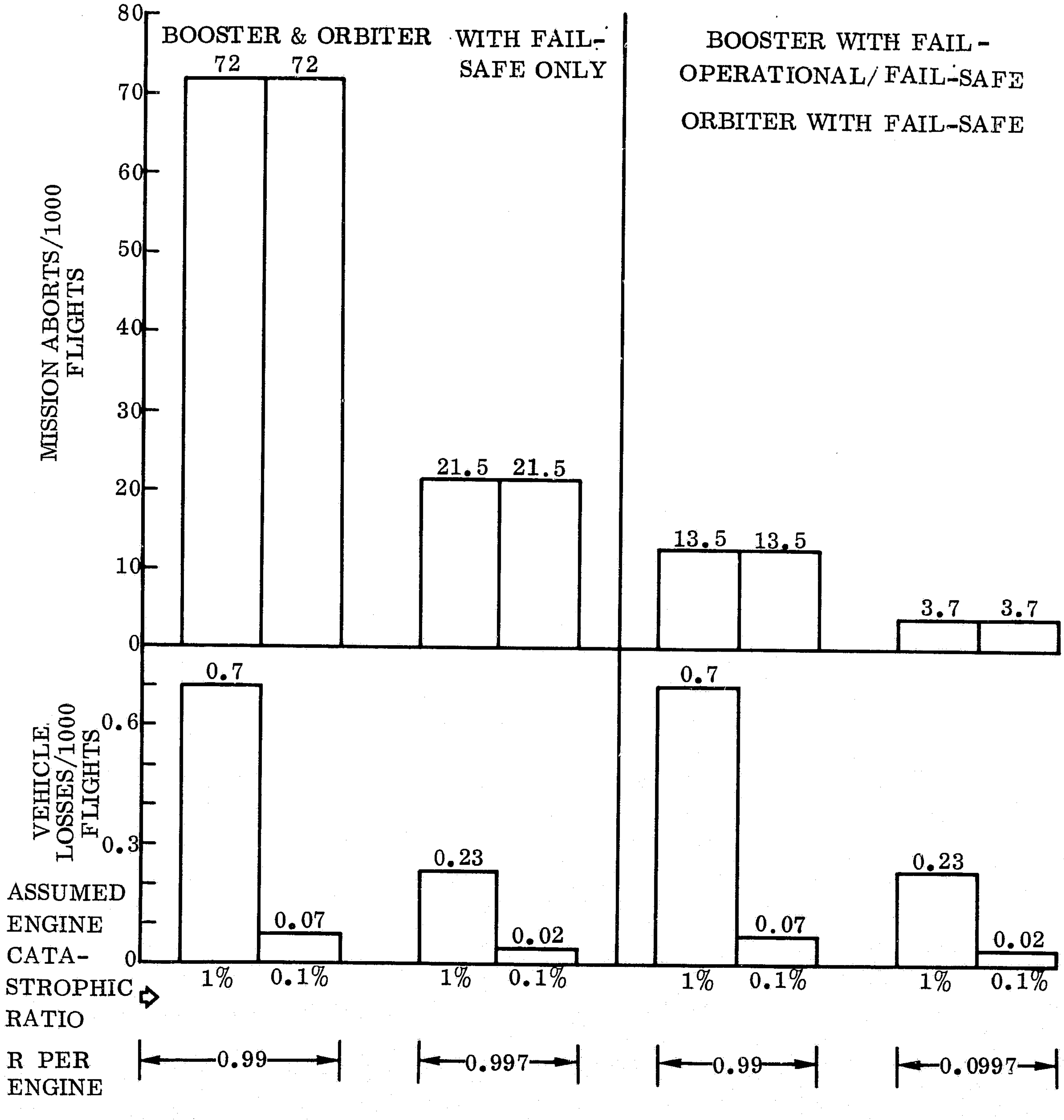


Figure 5-25. FR-3 Mission Aborts and Vehicle Losses for Different Engine Arrangements

Table 5-6. Losses/1000 Flights for 9-3-9 Engine Configuration

	Overthrust	GLOW (M lb)	As-sumed Catas. Ratio	Mission						Vehicles					
				R_{eng} = 0.99			0.997			0.99			0.997		
				B	O	T	B	O	T	B	O	T	B	O	T
F/S (B&O)	0	4.92	0.01	78	12	90	21	3.5	24.5	$\frac{0.8}{C}$	$\frac{0.1}{C}$	$\frac{0.9}{C}$	$\frac{0.2}{C}$	$\frac{0.03}{C}$	$\frac{0.23}{C}$
			0.001	78	12	90	21	3.5	24.5	$\frac{0.08}{C}$	$\frac{0.01}{C}$	$\frac{0.09}{C}$	$\frac{0.02}{C}$	$\frac{0.003}{C}$	$\frac{0.023}{C}$
F/O-F/S (B) F/S (O)	13%	5.04*	0.01	2.2	12	14.2	$\frac{0.22}{C}$	3.5	3.7	$\frac{0.8}{C}$	$\frac{0.1}{C}$	$\frac{0.9}{C}$	$\frac{0.2}{C}$	$\frac{0.03}{C}$	$\frac{0.23}{C}$
			0.001	2.2	12	14.2	$\frac{0.22}{C}$	3.5	3.7	$\frac{0.08}{C}$	$\frac{0.01}{C}$	$\frac{0.09}{C}$	$\frac{0.02}{C}$	$\frac{0.003}{C}$	$\frac{0.023}{C}$

* The 13% overthrust is beyond the presently designed engine limit for PU control needed for the fail-operational mode, and the constraint of using a 400,000-lb thrust engine module requires that 2 engines be added (1 for each booster) to obtain a 13% overthrust capability. A 10-3-10 configuration is required.

F/O-F/S = Fail-Operational/Fail-Safe
F/S = Fail Safe
GLOW = Gross Liftoff Weight
C = Loss Due to Engine Catastrophic Failure
B = Booster
O = Orbiter
T = Total

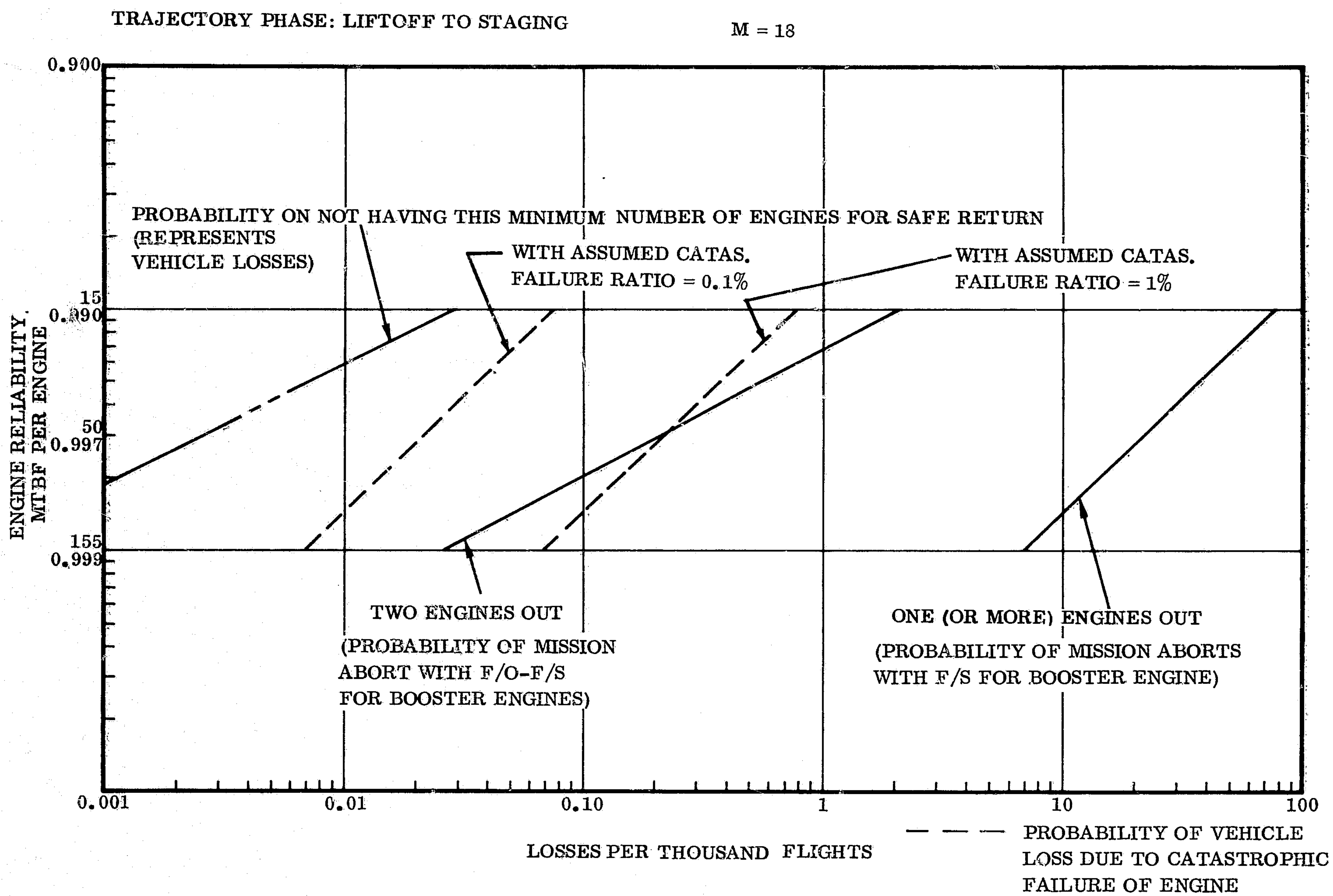


Figure 5-26. Engine Reliability Versus Losses/Aborts for 9-3-9 Engine Configuration

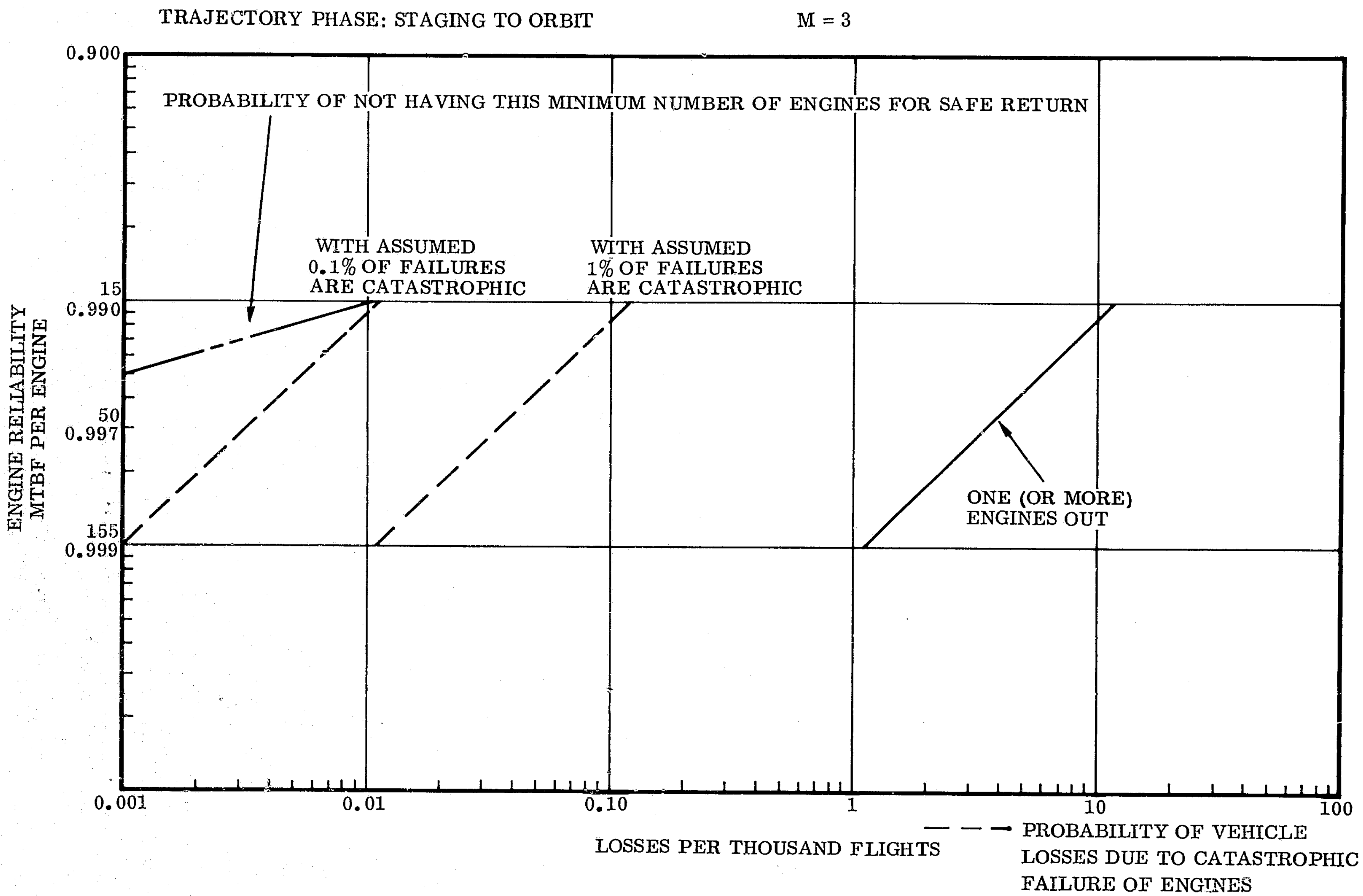


Figure 5-27. Engine Reliability Versus Losses/Aborts for 9-3-9 Engine Configuration

5.7.6.2 Evaluation of Results. Figure 5-28 is a graphic representation of Table 5-6. This figure shows a significant improvement in mission reliability with increasing application of F/O-F/S and improved engine reliability. Vehicle safety is not improved with increasing application of the F/O-F/S.

5.7.7 IMPLICATIONS OF PROPELLANT CROSSFEED FOR THREE-ELEMENT CONFIGURATIONS. The safety and operational implications of providing crossfeed to a three-element configuration appear in two areas.

a. Equalized Depletion for Boosters. Crossfeed enables booster propellants to be depleted equally so that dynamics at separation are symmetrical and a normal staging sequence can be executed following subsystem failure or engine failure during liftoff to staging.

b. Fail-Operational/Fail-Safe Application. For a multi-engine vehicle (say 9-3-9) with fail-operational/fail-safe provisions for booster engines, significant weight and cost savings can be made because the engine overthrust requirement is 6% rather than 13%. If provisions are made to achieve operational staging with a booster engine out fail-operational then:

   1. With crossfeed, the amount of overthrust $= \dfrac{1}{n_{b_{tot}} - 1} = \dfrac{1}{17} = 0.06 = 6\%$

      where

      $n_{b_{tot}}$ = total number of booster engines

      Minimum performance thrust to weight ratio is maintained and boosters reach staging with propellants equally depleted.

   2. Without crossfeed, the amount of overthrust required $= \dfrac{1}{n_{b_e} - 1} = \dfrac{1}{8} = 0.13 = 13\%$

      in order to achieve operational staging.

      where

      n_{b_e} = number of booster engines per element.

5.7.8 SUMMARY AND RECOMMENDATIONS

5.7.8.1 FR-3 Configuration. Table 5-7 summarizes the FR-3 rocket engine study.

Mission Losses. In order to approach a mission goal of 30 failures per 1000 flights, the 15-3 configuration with F/O-F/S for the boosters and F/S for the orbiter, with an engine reliability of 0.99, is required. Engine overthrust required is 7%.

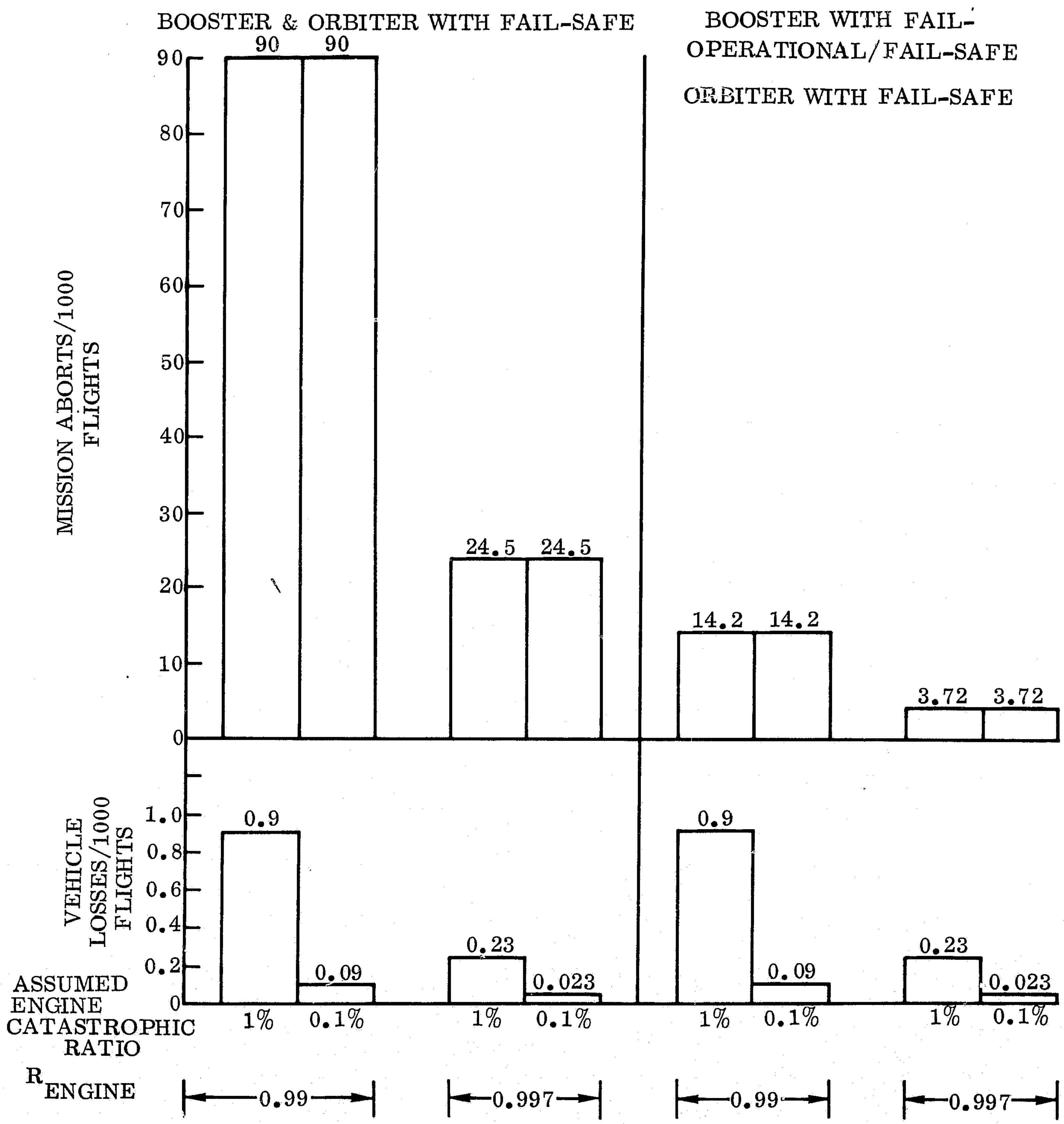


Figure 5-28. FR-4 Mission Aborts and Vehicle Losses for Various Engine Arrangements

Table 5-7. Summary of FR-3 Rocket Engine Study

Engine Configuration	Engine Reliability Required	Vehicle GLOW (M lb)	Engine Overthrust Capability (%)	Losses/1000 Flights All Causes: Mission, Calculated	Mission, Allowable	Vehicle (Safety): Calculated With Assumed Engine Catas. Ratio 1%	Calculated With Assumed Engine Catas. Ratio 0.1%	Vehicle (Safety): Allowable
15-3 F/S in (B & O)	0.99	4.32	0	92	30	1.33	0.70	1
	0.997	4.32	0	41	30	0.86	0.65	1
15-3 F/O-F/S in B; F/S in O	0.99*	4.32	7**	33	30	1.33	0.70	1
	0.997	4.32	7	24	30	0.86	0.65	1

* Recommended configuration (from the standpoint of safety, low mission losses, and attainable engine reliability of 0.99)

** The presently conceived rocket engine provides for up to a 7% overthrust with full PU control. Some refurbishment of the engine is required after use at the 107% thrust level.

Note: A value of 20 mission losses/1000 flights for causes other than engines is added to losses caused by engine failures to reflect total mission losses from all causes. A value of 0.63 vehicle losses/1000 flights for causes other than engine catastrophic failures is added to reflect total vehicle losses from all causes. The rationale for these values is based on the gross failure and mission termination analysis presented in Section 5.3.

F/O-F/S = Fail Operational-Fail Safe
F/S = Fail Safe
GLOW = Gross Liftoff Weight
B = Booster
O = Orbiter

Vehicle Losses. Vehicle losses are not significantly affected for any of the configurations shown. Reduction of the engine catastrophic ratio to less than 1% is required to meet the overall safety requirement.

Recommendations. The recommended configuration is the 15-3 with F/O-F/S in the booster and F/S in the orbiter. This configuration provides a significant decrease in mission losses approaching the allowable limit and achieves the safety criteria of 1/1000 losses.

An engine reliability of 0.99 (with a MTBF of 15 hours) with a 7% overthrust capability is recommended. With this provision the vehicle system can achieve staging with one engine out and is capable of safe return with two out. The orbiter is capable of safe return with one engine out after staging. On-orbit mission operations can be accomplished with one engine out and safe return to earth with two out.

Vehicle losses can be held within requirements with a catastrophic ratio ≤ 0.1% in a 15-3 arrangement and engine MTBF = 15 hours.

An engine reliability of 0.99 is a reasonable performance goal for engine development. Emphasis in the engine development program should be in reducing engine catastrophic failures to not over 0.1% of total engine failures.

5.7.8.2 FR-4 Configuration. Table 5-8 summarizes the FR-4 rocket engine study.

Mission Losses. In order to approach a mission goal of 30 failures per 1000 flights, the 9-3-9 configuration with F/O-F/S and 13% overthrust for the boosters and F/S for the orbiter, with the attendant engine reliability of 0.99, is required. Actually, in order to achieve a booster engine out fail-operational condition with 400,000 lb engine modules, a 10-3-10 configuration is needed with a gross liftoff weight increase of ≈120,000 pounds. Presently designed engine PU control capability cannot accommodate a 13% overthrust.

Vehicle Losses. Vehicle losses are not significantly affected for any of the configurations shown. Reduction of the engine catastrophic ratio to less than 1% is required to meet the overall safety requirement.

Recommendation. The recommended configuration is the 9-3-9 (actually 10-3-10) with F/O-F/S in the booster and F/S in the orbiter. This configuration provides a significant decrease in mission losses with a reasonable penalty. An engine reliability of 0.99 (with a MTBF of 15 hours) with a 13% overthrust capability is recommended. With this provision the vehicle system can achieve staging with one engine out and be capable of safe return with two out. The orbiter is capable of safe return with one engine out after staging. On-orbit mission operations can be accomplished with one engine out, and safe return to Earth with two out.

Table 5-8. Summary of FR-4 Rocket Engine Study

Engine Configuration	Engine Reliability Required	Vehicle GLOW (M lb)	Engine Overthrust Capability (%)	Losses/1000 Flights All Causes				
				Mission		Vehicle (Safety)		
				Calculated	Allowable	Calculated With Assumed Engine Catas. Ratio		Allowable
						1%	0.1%	
9-3-9 F/S in (B & O)	0.99	4.92	0	110	30	1.53	0.72	1
	0.997	4.92	0	45	30	0.86	0.65	1
9-3-9** F/O-F/S in B; F/S in O	0.99*	5.04	13	34	30	1.53	0.72	1
	0.997	5.04	13	24	30	0.86	0.65	1

* Recommended configuration (from the standpoint of safety, low mission losses, and attainable engine reliability of 0.99)

** The 13% overthrust is beyond the engine design limit for PU control needed for fail operational mode and the constraint of using a 400,000-lb thrust engine model requires that two engines be added (one for each booster) to obtain a 13% overthrust capability (a 10-3-10 configuration).

Note: A value of 20 mission losses/1000 flights for causes other than engines is added to losses caused by engine failures to express total mission losses from all causes. A value of 0.63 vehicle losses/1000 flights for causes other than engines is added to losses caused by engine catastrophic failure to reflect vehicle losses from all causes. The rationale for these values is based on the gross failure and mission termination analysis presented in Section 5.3.

F/O-F/S = Fail Operational-Fail Safe
F/S = Fail Safe
GLOW = Gross Liftoff Weight
B = Booster
O = Orbiter

vehicle losses can be held within requirements with catastrophic ratio ≤ 0.1% in a 9-3-9 (10-3-10) arrangement and engine MTBF = 15 hours.

5.7.8.3 Comparison of Fail Safe and Fail-Operational/Fail-Safe Applied to Rocket Engines. Figure 5-29 describes the fail-safe and fail-operational/fail-safe concepts. Table 5-9 shows aborted missions, and costs when fail-safe only is applied to the booster and the orbiter. The number of engines required, T/W, and probability of loss of engines are also shown. Table 5-10 shows the application of fail-operational/ fail-safe to the booster engines. Two approaches to providing performance T/W are shown: overthrust and throttling.

Table 5-11 shows weight and cost penalties for fail-operational/fail-safe for the overthrust and throttling approaches.

The engine refurbishment cost estimates shown are tentative. It is conceivable that the application of 7% overthrust could be accomplished with essentially no refurbishment. This is being determined by the engine manufacturer.

Table 5-12 shows a comparison of overthrust with the throttling approach and fail-safe versus fail-operational/fail-safe applied to the FR-3. If refurbishment costs can indeed be brought to zero then fail-operational/fail-safe will reduce cost of aborted mission $33,000,000 with 14 mission aborts. This compares with a cost of $166,000,000 and 72 mission aborts using the fail-safe concept. Both approaches (i.e., overthrust and throttling) results in reduced mission aborts and reduced cost when compared with utilizing the fail-safe concept.

Adding an engine to the orbiter to obtain fail-operational/fail-safe during staging to orbit is not cost effective for either overthrusting or throttling (data not shown on table). The weight penalty for adding an engine to the orbiter is on the order of 1/4 million pounds.

5.8 FINAL VEHICLE CHARACTERISTICS RELATIVE TO SAFETY, ABORT, AND MISSION SUCCESS

5.8.1 FR-3 VEHICLE CHARACTERISTICS. The safety and reliability characteristics of the FR-3 are summarized in Table 5-13. Significant points are:

a. The vehicle demonstrates a predicted high safety (intact abort success) of 0.999 (1 loss/1000 flights) if engine catastrophic failures are ≤ 0.1%.

b. The mission reliability goal for the FR-3 is 0.970 (30 aborts/1000 flights). The predicted mission success using an engine reliability of 0.99 is ≈0.970 and using an engine reliability of 0.997, a 0.976 mission success probability is achieved.

c. All mechanical/electrical subsystems provide F/O-F/S capability.

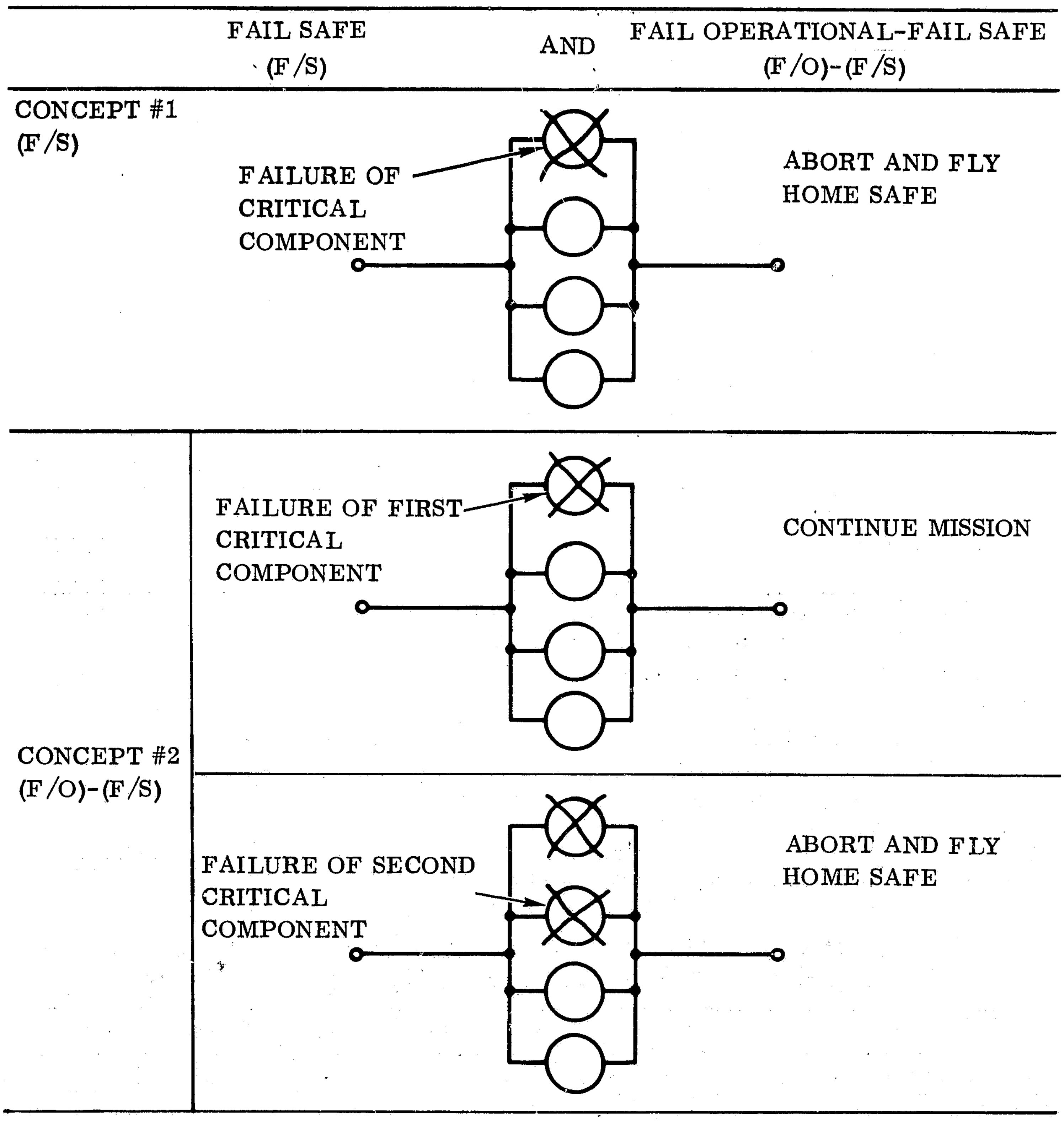


Figure 5-29. Fail Safe and Fail Operational-Fail Safe Concepts

Table 5-9. Main Propulsion — F-S Versus F-O/F-S With F-S Only Applied to Booster and Orbiter

	Booster	Orbiter
Number of Rocket Engines	15	3
Performance T/W	1.386	1.527
Number of Engines Required for T/W	15	3
Probability of Not Having Required Engines (i.e., one engine out)	60/1000 flights	12/1000 flights
Aborted Missions/1000 Flights	60	12
Δ$ Aborted Missions	$138M	$28M
	$166 M Total	

Table 5-10. Main Propulsion — F-S Versus F-O/F-S With F-O/F-S Applied to Booster Only

	Booster*	Orbiter	Booster**	Orbiter
Number of Rocket Engines	15	3	16	3
Performance T/W	1.386 (with 14)	1.527 (with 3)	1.386 (with 15)	1.527 (with 3)
Probability of One Engine Out	60	12	≈60	12
Probability of Not Having Required Engines (i.e., two out on booster, 1 out on orbiter)	2	12	2	12
Aborted Missions/1000 Flights	2	12	2	12
Δ$ Aborted Missions	$5M	$28M	$5M	$28M
Overthrust (Overspeed) Required	7%	-	-	-
Throttling Required	-	-	0.93	-

*With Overthrust (Overspeed); ** Throttled (Add One Engine)

Table 5-11. Cost and Weight of Overthrust and Throttling for Fail-Operational/ Fail-Safe for Booster (Main Propulsion)

	Booster*	Orbiter	Booster**	Orbiter
Engine Refurbishment Cost After Use in 7% Overthrust (overspeed)				
@ 25% of Initial Cost	$810M	-	-	-
@ 5%	$162M	-	-	-
Weight Added to Engine, Thrust, Structure, etc. (GLOW), lb	-	-	54K	
$ Added Weight	-	-	$32M	-
Added Engine	-	-	66M	
Thrust Structure, Propellant Feed, etc.	-	-	12M	
			110M	

*With Overthrust; **With Throttling

Table 5-12. Overthrust/Throttling Cost Comparison

Δ$ With Overthrust			
Aborted Missions	$ 33M	33M	33M
Engine Refurbishment	@25 810M	@5% 62M	0% 0
	843M	195M	33M
Δ$ With Throttling (Added Engine)			
Aborted Missions	33M		
Added Weight + One Engine, etc.	110M		
	143 M		

	Fail/Safe, Booster and Orbiter	Fail-Operational/Fail-Safe, Booster & Fail Safe Orbiter	
		With Overthrust	With Throttling
Δ$	$166M	33M or 843M	143M
Aborted Missions	72	14	14

d. Once-around abort capability is achieved with up to three engines out during boost 1. The same capability is achieved with up to two engines out during boost 2.

5.8.2 FR-3 VEHICLE OPERATIONS AND ABORT

Operations (Mission Completion). The vehicle cluster will reach staging with one booster engine out. The minimum performance thrust-to-weight ratio is maintained by providing a 15-3 engine arrangement with engine overthrust capability ≈7% or added booster engine in a 16-3 arrangement. In the 15-3 arrangement, when a booster engine fails the remaining 14 engines are stepped up 7% to give same initial total thrust. Some refurbishment of engines is required after this operation. For the 16-3 arrangement, the 16 engines are normally throttled to 93%. Upon loss of an engine, the remaining 15 are advanced to 100%.

On-orbit operational maneuvers for the three-engine orbiter are accomplished with one or even two engines out. The probability of loss of three engines in the orbiter is a negligible number; however, if this should happen, it is possible to de-orbit using the attitude control system. In summary then, the orbiter has F/O-F/S capability for the on-orbit maneuver.

FR-3 Abort. Abort procedures for the FR-3 are essentially the same as for the FR-1 vehicle. The number of mission aborts required is low because of the F/O-F/S provision in the recommended vehicle which makes possible a 0.97 mission

Table 5-13. FR-3 Basic Vehicle Characteristics Relative to Safety, Abort, and Mission Success

Payload	50,000 lb
Mission Time	7 days
GLOW (gross liftoff weight)	*4.32×10^6 lb
Rocket Engine Configuration	**15-3 (3 orbiter engines)
$T/W_{L.O.}$ (one rocket engine out)	1.38 (with F/O-F/S - 7% overthrust)
$T/W_{L.O.}$ (two rocket engines out)	1.2 ($T/W_{L.O.}$ required for O/A < 1.2)
$T/W_{Staging}$ (one rocket engine out)	1.0 ($T/W_{staging}$ required for O/A < 1.0)
F/O-F/S Capability for rocket engines	Yes (for booster) No (orbiter)
F/O-F/S Capability for subsystems	Yes
"Once Around" Abort Capability With	
1 Rocket engine out (boost 1)*	Yes
2 Rocket engines out (boost 1)	Yes
3 Rocket engines out (boost 1)	Yes at $t \approx 20$ sec
1 Rocket engine out (boost 2)	Yes
2 Rocket engines out (boost 2)	Yes at $t \approx 100$ sec into boost 2
Intact Abort Capability	Yes
Propellant Crossfeed	No
Propellant Dump Provision	No

Initial Reliability Goals		Achieved Reliability Goals (Predicted)	
		For $R_{eng} = 0.99$	For $R_{eng} = 0.997$
Safety (Intact Abort Success)	0.999 (1 loss/ 1000 flights)	0.999 (1 loss/1000 flights) >0.999 (<1 loss/1000 flights)	0.999 (1 loss/1000 flights) >0.999 (<1 loss/1000 flights)
Mission Success	0.970 (30 aborts/ 1000 flights)	≈0.97 (33 aborts/1000 flights)	>0.976 (24 aborts/1000 flights)

* 7% overthrust for rocket engines.

** Alternate engine provision-GLOW (4.38×10^6 lb - with engines throttled in booster — added engines in a 16-3 engine arrangement)

reliability resulting in 33 mission aborts/1000 flights, approximately 4/year based on 1000 flights/10 years.

During liftoff to staging, loss of two engines will be cause for abort. The abort options for the FR-3 are also applicable to FR-1 (i.e., "once around" and intact abort).

After staging, the orbiter will achieve a once-around the Earth abort maneuver and land at the launch site if an engine fails during staging to orbit.

5.8.3 FR-4 VEHICLE CHARACTERISTICS. The safety and reliability characteristics of the FR-4 are summarized in Table 5-14. Significant points are:

a. Predicted mission reliability using an engine reliability of 0.99 is ≈ 0.89. Using an engine reliability of 0.997, mission success is improved to 0.95.

b. Safety (intact abort success) goal is achieved with either engine reliability (0.99 or 0.997) if engine catastrophic failures are $\leq 0.1\%$.

c. All mechanical/electrical subsystems have the fail-operational/fail-safe capability.

d. "Once around" abort capability is achieved with up to three rocket engines out during boost 1. The same capability is achieved with up to two rocket engines out during boost 2.

F/O-F/S for FR-4 boosters can be provided with a 10-3-10 arrangement as shown in Table 5-14 with weight penalty as shown.

5.8.4 FR-4 OPERATIONS AND ABORT

Operations (Mission Completion). The vehicle cluster with the 9-3-9 engine arrangement cannot reach staging with one booster engine out. Loss of a booster engine will be cause to abort the mission. For the alternate engine arrangement (10-3-10) engines are normally throttled to $\approx 91\%$. In event of failure of an engine (in one booster) remaining 9 are advanced to 100% and the normal staging point is achieved.

On-orbit operational maneuvers for the three-engine orbiter are the same as for the FR-3.

FR-4 Abort. Abort procedures for the FR-4 are essentially the same as for the FR-1 vehicle. The number of aborts is high because there is no fail-operational/fail-safe provision. The abort options for the FR-1 are also applicable to the FR-4 (i.e., "once around" and intact abort).

After staging, the orbiter will achieve a once-around the Earth abort maneuver and land at the launch site if an engine fails during staging to orbit.

Table 5-14. FR-4 Basic Vehicle Characteristics Relative to Safety, Abort, and Mission Success

Payload	50,000 lb
Mission Time	7 days
GLOW (gross liftoff weight)	*4.92×10^6 lb
Rocket Engine Configuration	**9-3-9 (3 orbiter engines)
$T/W_{L.O.}$ (one rocket engine out)	1.38 ($T/W_{L.O.}$ = 1.47 is required to reach staging)
$T/W_{L.O.}$ (two rocket engines out)	1.30 ($T/W_{L.O.}$ required for "once around" <1.3)
$T/W_{Staging}$ (one rocket engine out)	0.81 ($T/W_{staging}$ required for "once around" < 0.8)
F/O-F/S Capability for rocket engine	No (except for alternate)**
F/O-F/S Capability for subsystems	Yes
"Once Around" Abort Capability With	
1 Rocket engine out (boost 1)	Yes
2 Rocket engines out (boost 1)	Yes
3 Rocket engines out (boost 1)	Yes, at t ≈ 20 sec
1 Rocket engine out (boost 2)	Yes
2 Rocket engines out (boost 2)	Yes, at t ≈ 100 sec into boost 2
Intact Abort Capability	Yes
Propellant Crossfeed	No
Propellant Dump Provision	No

	Initial Reliability Goals	Achieved Reliability Goals (Predicted)	
		For R_{eng} = 0.99	For R_{eng} = 0.977
Safety (Intact Abort Success)	0.999 (1 loss/ 1000 flights)	*>0.999 (<1 loss/1000 flights)	*>0.999 (<1 loss/1000 flights)
Mission Success	0.970 (30 aborts/ 1000 flights)	*0.89 (110 aborts/1000 flights)	*0.95 (45 aborts/1000 flights)

** Alternate engine provision - GLOW 5.04×10^6 with F/O-F/S for boosters in a 10-3-10 engine arrangement gives: safety (intact abort success) = 0.999 (1 loss/ 1000 flights), mission success = 0.97 (34 aborts/1000 flights).

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